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PRELIMINARY MISSION DESIGNS FOR JUPITER ORBITER MISSIONS

by Byron L. Swenson, Edward L. Tindle, and Larry A. Manning

Advanced Concepts and Missions Division
Office of Aeronautics and Space Technology
Moffett Field, Calif. 94035

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16. Abstract

Preliminary designs for unmanned orbital exploration missions to Jupiter have been examined. This report summarizes the operational options and systems requirements consistent with the major scientific goals of the mission. In general, each mission design discussed provides repeated measurements of the interaction of Jupiter with the solar media; encounters at least two Galilean satellites with multiple encounters with at least one satellite at distances which allow photography with resolutions of at least 10 km; provides at least 10 orbital maps of the field and particle environment surrounding Jupiter; and provides synoptic observations of Jupiter over a range of wavelengths and various degrees of photographic coverage with resolutions of 300 to 30 km.

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SUMMARY

Preliminary designs for unmanned orbital exploration missions to Jupiter have been examined. This report summarizes the operational options and systems requirements consistent with the major scientific goals of the mission. In general, each mission design discussed provides repeated measurements of the interaction of Jupiter with the solar media; encounters at least two Galilean satellites with multiple encounters with at least one satellite at distances which allow photography with resolutions of at least 10 km; provides at least 10 orbital maps of the field and particle environment surrounding Jupiter; and provides synoptic observations of Jupiter over a range of wavelengths and various degrees of photographic coverage with resolutions of 300 to 30 km.

INTRODUCTION

Orbiting spacecraft about the planet Jupiter are receiving considerable attention as an attractive means of answering some of the basic scientific questions relating to that body, the near Jovian space, and the satellite system surrounding Jupiter. The observational objectives of such a spacecraft are numerous and, of course, result in a set of constraints outside of which the interpretation of the observables becomes relatively more difficult and the observations have significantly less value. This paper examines the orbital selection and operations problem in light of these constraints to determine if the observational objectives of such a mission can be satisfied in general and, in particular, whether a large number of the observations can be made with a single spacecraft.

The discussion moves from a definition of the observational requirements for an orbiter mission to general consideration of the orbit selection process. A complete specification of the characteristics of the highly desirable orbital mission designs is then presented. Some of the problem areas and solutions associated with these mission designs are then discussed. Finally, several alternative mission designs and strategies are examined that have less stringent systems requirements but are less desirable from a scientific point of view.

The actual mission designs chosen are not too important. It will be demonstrated that considerable flexibility exists in the mission design. Thus, the actual missions described herein can be modified considerably as the science objectives and instrument capabilities are examined in greater detail. The intent here is to lay some of the groundwork for the final mission design by indicating some of the many options available.

OBSERVATIONAL REQUIREMENTS

The observational requirements for an orbiter mission to Jupiter can be divided into four general areas of scientific interest. These areas relate to the observation and understanding of:

1. The interaction of Jupiter and the solar media.

2. The major regular satellites of Jupiter and their interactions with the Jovian magnetosphere.

3. The near planetary environment and phenomena.

4. Planetological phenomena including the dynamics and structure of the atmosphere.

Some of the observational rationale and requirements in each of these areas are summarized in reference 1. Additional details can be found in references 2 and 3. In each area, emphasis is placed on some understanding of the associated phenomena. A good understanding in one area will not come about without a good understanding in the others. However, it is convenient to divide the total science requirements in this manner in order to make the orbit selection problem tractable.

Associated with each observational objective in each area is a set of observational constraints. In most cases, the observational constraints are not defined sharply. For example, a discrete solar zenith or phase angle range may be specified for a given observation, but this does not imply that the value of the observation is a step function at each of the limits. Rather, experience (or in some cases theory) has indicated that the interpretation of the observation will be enhanced if phase angle is approximately within this range.

Interaction Region Experiments

To accomplish the interaction region experiments, it is necessary, of course, to traverse the magnetopause and shock wave. Estimates indicate that this region is located between 80 to 100 R_J (Jupiter radii) from the center of Jupiter in the direction of the Sun. It is also desirable to make measurements within about 45° of the Jupiter-Sun line, but measurements up to 90° away from that line would be of considerable value. It is also desirable to traverse this region at least three to five times.

Jovian Satellite Observations

Images with a surface resolution better than 10 km are needed for meaningful study of the morphology of the surface of the major Jovian satellites. Thus, with an imaging system with a resolution capability of, say, 200 μ rad, it is necessary to encounter the satellite within about 50,000 km or about 0.7 R_J'to achieve the desired surface resolution. It is also desirable to view the satellites over a wide range of phase angles and to have multiple encounters with each satellite. In addition, there is strong evidence that the satellite Io modulates the radio emission of Jupiter through its interaction with the Jovian magnetic field. Measurements made within the magnetic flux

tube intersecting Io would lead to a greater understanding of this phenomenon. For these measurements, therefore, it is desirable to encounter Io either over its north or south pole.

Measurements of Fields and Particles

To sample the near Jovian environment, it is for the most part necessary to traverse the regions of the near planetary space of interest. The magnetic fields and trapped energetic particle belts encompass the region of space from the upper fringe of the atmosphere out to about 6 R_J for the trapped particle belts and out to the interaction region (say, 80 R_J) for the magnetic field. It is desired to sample this environment for at least 10 orbits.

Planetology Observations

The fourth area of planetology, which includes the dynamics and structure of the atmosphere, has the largest variety of phase angle and distance requirements. In general, the first objective is to obtain full disk imagery with a resolution at least an order of magnitude better than the average images obtainable from Earth-based equipment. A typical resolution obtained from Earth is about 3000 km. In addition, a much greater range of phase angle for full disk images is desired than can be obtained from Earth. Current Earth-based images are limited to a range of phase angles from 0° to 12°. After obtaining full disk images it is desired to obtain nested higher resolution images for detailed cloud motion studies down to a resolution of better than 30 km. Time-lapse imagery is also highly desirable.

Observations in the UV, IR, and microwave regions of the spectrum are also desirable for a variety of scientific reasons (refs. 1, 2, and 3). Dayside and nightside observations, and measurements for at least 10 orbits, are desired.

ORBIT SELECTION CONSIDERATIONS

The selection of the appropriate orbit or orbits for a Jupiter orbiting mission involves balancing the above scientific requirements and desires against the capability of the spacecraft system and constraints of the overall mission. The capability of the spacecraft system is measured in terms of its propulsion capacity, its tolerance to the environment (primarily trapped radiation), and the view angle limits available. The mission constraints result from the launch vehicle used, the particular launch opportunity chosen and the transfer trajectories available during that opportunity, and the accuracy with which that mission can be flown.

A preliminary examination of the orbit selection problem indicates that within reasonable propulsion capabilities very little freedom is available in the choice of the orientation of posigrade orbits about Jupiter. In general, possible posigrade orbits will be highly elliptical and will be oriented with periapsis very near the sunset terminator; thus, the line of apsides will be approximately at right angles to the Jupiter-Sun line. The only real choices to be made in orbit selection involve the time of periapsis passage, the inclination of the orbit, and, depending on propulsive capability, the size of the orbit (periapsis and apoapsis radius).

The most stringent of all the scientific requirements from an orbit selection point of view is the requirement to have close multiple encounters with some of the major satellites. For the most part the other observations can be made just as well on an orbit sized for satellite encounters as on any other orbit. Thus, the orbit selection strategy given here is dominated by the requirement to encounter satellites but does not ignore the importance of the other measurements.

Orbits having multiple satellite encounters are possible at Jupiter due to the commensurable motion of Io, Europa, and Ganymede, three of the four large Galilean satellites. Niehoff (ref. 4) has analyzed orbits that will encounter all three of these satellites in a single orbit followed by similar encounters on subsequent orbits. Basically the motion of Io, Europa, and Ganymede about their very nearly circular equatorial orbits is such that the ratio of their respective periods is 1:2:4. This motion results in an alinement known as syzygy (fig. 1), in which Io and Ganymede are alined on one side of Jupiter while Europa is nearly exactly (within less than 0.5°) 180° away from this alinement. This alinement occurs every 7.05 days. Because the motions of these three satellites are not quite commensurable, however, the alinement drifts inertially in a retrograde fashion about 5.2° between one syzygy and the next. Since syzygy occurs about every 7 days, this alinement can be considered as a starting condition in the calculation of encounter orbits.

In reference 4, the size (periapsis and apoapsis radius) of orbits that will encounter satellites has been calculated and depends on the sequence in which the satellites are intercepted. Niehoff indicates that mode 3 – an encounter sequence involving intercepts with Io and Ganymede after periapsis passage and Europa just before periapsis passage – and the mirror image, mode 7 – Ganymede and Io before periapsis passage and Europa after passage – are the most practical encounter modes considering possible radiation hazards. For each mode, particular orbits with periods that are integer multiples of 7.111 days (a compromise period near that of Ganymede) will encounter each of the three satellites in a single orbit. However, due to the regression of the line of syzygy, good encounters will occur only for a few orbits, depending on the orbit period. The size characteristics of these orbits are listed in table 1. Table 1 also gives the required relative orientation of syzygy and periapsis and the required time of periapsis passage relative to syzygy as calculated in appendix A. For clarity, the orientation and timing requirements for both mode 3 and mode 7 encounters for the 14.222-day orbit are shown in figure 2.

TABLE 1.– MODE 3 ORBIT CHARACTERISTICS

Orbit period, days	7.111	14.222	21.333
Periapsis radius, R_J	2.391	2.290	2.255
Apoapsis radius, R_J	27.483	45.131	59.884
True anomaly of syzygy, deg	118.74	117.13	116.53
Time of periapsis relative to syzygy, day	–0.263	–0.249	–0.243

To determine whether either of these alinements occurs within a short time after the arrival date for particular missions, example missions were analyzed for the 1976, 1977, and 1981 opportunities. The 1976 and 1977 opportunities represent the earliest opportunities for an orbiter mission and have the most stringent energy requirements. The 1981 mission is representative of the lowest energy opportunities for Jupiter orbiter missions. The alinements of the line of apsides of the encounter orbit relative to syzygy are shown for these three launch opportunities in figures 3(a)-3(c). The solid line indicates the right ascension (i.e., longitude in a planet-centered inertial coordinate system with the X axis directed in the direction of Jupiter's vernal equinox and with the Z axis in the direction of the north pole) of periapsis for an on-periapsis insertion into an

elliptic orbit with periapsis radius equal to 2.29 R_J (required for the 14.222-day orbit) as a function of arrival date (given in the Julian calendar). The corresponding trip times to Jupiter are also shown by the tick marks. In addition, the right ascension of the syzygy alinements (defined in the direction of Io and Ganymede) for the various arrival dates is shown by the dashed line. It can be seen that syzygy regresses inertially 180° in about 250 days.

From figure 2 it can be seen that the mode 3 encounters require periapsis alinement about 117° back from syzygy. This condition is indicated in figure 3 by the dash-dot line parallel to the syzygy line labeled "mode 3 periapsis alinement." Similarly, mode 7 encounters require periapsis to be alined 117° ahead of syzygy; this condition is indicated by the dash-dot line labeled "mode 7 periapsis alinement." It is now obvious from the examination of figures 3(a) and 3(b) that the arrival conditions in 1976 and 1977 are not correct for mode 3 encounters and that one must wait 200 to 500 days, depending on arrival date, before syzygy regresses back to the proper orientation with respect to the inertially fixed orbit. It is also apparent, however, that mode 7 encounters will occur 3 to 4 weeks after arrival along a 900-day transfer trajectory. If the trip time is shortened, the wait time for encounters is increased by a like amount. In 1981 (fig. 3(c)) just the opposite situation exists in that mode 3 encounters occur somewhat after arrival and mode 7 encounters require rather long wait times.

The appropriate mode 3 or 7 periapsis locations are shown in figure 3 as a continuous line with respect to arrival date. Actually these alinements are a set of discrete points separated by about 7 days. The choice of the target syzygy for the 1976, 1977, and 1981 missions is shown in figures 4(a)-4(c), in which the details of the arrival conditions and the target encounter opportunities are given on a magnified right ascension scale. In each case the data points on the right-hand side indicate three alinements of periapsis for three consecutive syzygy dates that result in mode 7 encounters (mode 3 in 1981). These periapsis alinements are in the range of periapsis alinements resulting from on-periapsis orbital insertions from short to long trip-time type I transfers to Jupiter.

It is, of course, possible to perform an off-periapsis insertion into orbit and thus increase the freedom of choice of the target syzygy date. This can only be done, however, at a considerable propulsion penalty, as shown in figure 5. The off-periapsis insertion penalty is shown for three values of periapsis radius from 1.1 to 4 R_J. It can be seen that as little as 3° of periapsis rotation requires a propulsive penalty of about 100 m/sec. For this reason it appears desirable to limit periapsis rotation to about 2°.

The choice between the three syzygy dates shown in figure 4 for each launch opportunity is dictated by the characteristics of the transfer trajectories. The short trip time transfers of 650 to 750 days have rather high injection energy requirements, C_3, and also have rather high hyperbolic approach speeds at Jupiter, which result in increased orbital insertion velocity requirements. The much longer 850- to 950-day transfers have the lowest hyperbolic approach speeds but again have rather high injection energy requirements and can have high Earth departure declinations. Transfers of 750 to 850 days are nearly minimum energy trajectories, and at the same time the hyperbolic approach speeds are a reasonable compromise. Considering these facts and the nominal velocity penalty associated with periapsis rotations of ±2°, the middle syzygy date was chosen as the target syzygy date for each mission opportunity. This choice allows a wide range of possible trip times between about 750 and 850 days and corresponding arrival dates over about a 100-day period without violating the ±2° periapsis rotation constraint.

For example (fig. 4(a)), an on-periapsis arrival after an 815-day trip during the 1976 launch opportunity results in an orbit that is properly alined for close satellite encounters near the syzygy occurring on Julian day 2443933 (Feb. 28, 1979). The arrival date for this transfer is 2443815 (Nov. 2, 1978), and thus the best satellite encounters occur about 118 days after arrival. This wait time for encounters can be changed by taking, say, a 760-day trip arriving at 2443750 and inserting in an off-periapsis manner to rotate periapsis -2° to match the alinement required for the target syzygy on 2443933. The wait time would then be 183 days. The wait time for encounters can be shortened by taking a 860-day trip arriving at 2443858 and inserting with a $+2^\circ$ periapsis rotation. The wait time is then 75 days.

Examination of figures 4(b) and 4(c) shows with periapsis rotations within the range of $\pm 2^\circ$ that the orbital wait times for encounters for the 1977 and 1981 launch periods range from 113 to 203 days and 22 to 180 days, respectively.

Some orbital wait time before encounters is desirable for three reasons. First, some time is required to track and trim out insertion errors. Second, the requirements for the interaction region experiments dictate initially that a very loose orbit be established in order to make several passes through the bow shock at about 80 R_J. Finally, the results of reference 4 and the present results indicate that close encounters with the satellites occur for about three to five orbits centered about the target syzygy date. Thus, for example, if a 14.222-day (2.29X45.131 R_J) encounter orbit is chosen, then it is required to be at periapsis 0.249 day after syzygy for the best mode 7 encounter (see table 1). To achieve the maximum number of close encounters, it is necessary to enter that orbit at least two orbit periods before that time (28.444 days). Finally, to achieve at least five passes through the bow shock it is required to have initially two orbits of about 2.29X100 R_J (this provides one bow shock pass during hyperbolic approach and two passes on each orbit). The time spent in these two orbits is about 90 days. Therefore, the desired total orbit wait time to meet the various science requirements is, in this case, approximately 118 days.

MISSION SPECIFICATION

Using this rationale, the timing requirements to satisfy the dictates of the orbital science for missions during the 1976, 1977, and 1981 launch opportunities are indicated in table 2. Each

TABLE 2.– TIMING REQUIREMENTS FOR JUPITER ORBITER MISSIONS

	1976 mission (mode 7)		1977 mission (mode 7)		1981 mission (mode 3)	
Target syzygy date	2443933.669	Mar. 1, 1979	2444370.827	May 11, 1980	2445830.368	May 9, 1984
Timing requirement for encounter	+.249		+.249		−.249	
Periapsis date for best encounter	2443933.918	Mar. 1, 1979	2444371.076	May 11, 1980	2445830.119	May 9, 1984
Two orbit periods	−28.444		−28.444		−28.444	
Date of entrance into 2.29X45.131 R_J orbit	2443905.474	Jan. 31, 1979	2444342.632	Apr. 13, 1980	2445801.675	Apr. 11, 1984
Two interaction region orbits (2.29X100 R_J)	−90.154		−90.154		−90.154	
Arrival date	2443815.320	Nov. 2, 1978	2444252.478	Jan. 13, 1980	2445711.521	Jan. 12, 1984

mission defined has a fixed time of arrival at Jupiter. The variations in departure and arrival conditions over a typical launch window for those fixed arrival dates are shown in figures 6(a)-6(c).

For a 20-day launch window, figure 6 indicates that departure energy requirements C_3 for 1976 and 1977 are somewhat larger than that required in 1981. In 1976, the maximum C_3 required for a 20-day launch window is 93 km^2/sec^2 as compared with 105 km^2/sec^2 in 1977 and 90 km^2/sec^2 in 1981. The implications of these required launch energies on maximum launched payload for Titan-Centaur launch vehicles with various upper stages are shown in figure 7. Use of the spin-stabilized TE–364–4 upper stage results in maximum injected spacecraft weights of 1060 kg in 1976, 880 kg in 1977, and 1120 kg in 1981. Use of the three-axis stabilized Burner II upper stage results in a decrease of about 100 kg in the maximum injected spacecraft weight, regardless of launch energy.

The other trajectory parameter of principal interest for departure is the departure declination of the hyperbolic asymptote (fig. 6). The departure declinations available for launch along the Eastern Test Range (ETR) are limited by range safety azimuth constraints and by the allowable coast time in the departure parking orbit. The Centaur stage is currently limited in orbital coast time to approximately 24 min (ref. 5). The maximum departure declinations shown in figure 6 range from about 53° for 1977 and 36° in 1976 to a low value of 6.5° in 1981. Note that the departure declinations in 1976 and 1977 can be reduced significantly at some penalty in C_3 by moving the launch window to later departure dates. The implications of the high departure declinations on orbit coast time for various launch azimuths within the range safety constraints of the ETR are indicated in figure 8. The data are for a departure excess speed V_∞ of 0.3 emos, which is typical for the transfer trajectories under consideration. It can be seen that to achieve departure declinations greater than 30° requires coast times in excess of 40 min. Thus, some modification of the Centaur or its orbital operating procedure is required for missions launched in 1976 or 1977. Such a modification is not considered difficult.

Figures 6(a)-6(c) also give the variation of arrival conditions across the launch window. The hyperbolic excess speed V_∞ approaching Jupiter is of fundamental importance, since it and the orbit size determine the insertion velocity requirements. The hyperbolic excess speeds for the chosen 1976 and 1977 missions are nearly 6 km/sec, while that for the 1981 mission is about 7.6 km/sec. The insertion velocity requirement to achieve the nominal 2.29X45.131 R_J orbit is shown as a function of V_∞ in figure 9. The insertion velocity requirements for 1976 and 1977 are seen to be slightly over 1400 m/sec as compared with 1700 m/sec for the 1981 mission.

The arrival declination of the hyperbolic asymptote relative to the Jupiter equator, also shown in figures 6(a)-6(c), is directly related to the plane change requirement in order to achieve an equatorial orbit. Since all the orbit planes of the Galilean satellites are nearly in the plane of Jupiter's equator, it is obvious that the satellite encounters will be the best for an equatorial orbiting spacecraft. The maximum approach declinations during a 20-day launch window range from about −13° in 1977 and about −10° in 1976 to only 3.5° for the 1981 mission. It should be noted that the arrival declinations in 1976 and 1977 can be lowered substantially at some penalty in launch energy C_3 by shifting slightly the launch window to later launch dates.

The plane-change ΔV requirements for these declinations are shown in figure 10. The data in this figure have been calculated assuming that the plane-change maneuver is made in an optimum manner near the 100 R_J apoapsis of either the first or second interaction region orbit. It can be seen

from these results that for an orbit with a periapsis radius of 2.29 R_J about 740 m/sec is required to plane change into an equatorial orbit if the worst case declination of −13° occurs at arrival for the 1977 opportunity. This is compared to a worst case plane-change requirement of 540 m/sec and 180 m/sec for the launch windows for the 1976 and 1981 opportunities. The effect on the required plane-change maneuver of changing periapsis radius to 1.1 or 4 R_J is indicated by the labeled dashed lines. It can be seen that there is very little effect of periapsis radius on the optimum plane-change maneuver except for high declinations.

For both the 1976 and 1977 launch opportunities, the high departure and arrival declinations near the start of the 20-day launch window are caused by the proximity of that launch date to the boundary between Type I and Type II transfers. This boundary, called the "ridge," is where the heliocentric transfer angle is nearly 180°. These characteristics of single plane transfers can be eliminated through the use of broken plane transfers. For example, a broken plane transfer near the start of the launch window in 1976 can reduce departure declination to about 16° and the arrival declination to about −5° for a cost of about 155 m/sec midcourse maneuver about 150 days after launch. This broken plane transfer does not increase either the launch energy C_3 or the arrival hyperbolic excess speed V_∞. This maneuver can be accomplished by the orbit insertion and maneuvering system, resulting in a net saving of about 200 m/sec in the total requirement (i.e., midcourse, insertion, plane change, and deboost) for that system.

The total ΔV budgets for the 14.222-day three-satellite encounter orbits being considered here for the three launch opportunities are indicated in table 3. The first item is the maximum on-periapsis insertion ΔV required to establish the interaction region orbit (2.29×100 R_J). The second item indicates the penalty to rotate periapsis during the insertion maneuver to aline it properly for encounters. Such rotation is not required for the 1976 and 1981 opportunities; for the 1977 opportunity, however, a 60 m/sec penalty is required to rotate periapsis 2° for the proper alinement. The third item in the ΔV budget is the worst plane-change requirement to achieve an equatorial orbit across the 20-day launch window. The fourth item is the subsequent deboost requirement from the interaction region orbit into the satellite encounter orbit (2.29×45.131 R_J) applied at periapsis after two interaction region orbits. Finally, the last item indicates the ΔV requirement to trim periodically the orbit period of the encounter orbits to compensate for the perturbation effects of the satellites; this contribution to the ΔV budget was estimated from the results of reference 4.

TABLE 3.– ΔV BUDGET – THREE-SATELLITE ENCOUNTER ORBIT

	1976	1977	1981
Insert into 2.29×100 R_J orbit	900 m/sec	900 m/sec	1160 m/sec
Periapsis rotation penalty at insertion	0	60	0
Plane-change to equatorial orbit	540	740	180
Deboost into 2.29×45.131 R_J orbit	520	520	520
Satellite perturbation trim	60	60	60
TOTAL	2020 m/sec	2280 m/sec	1920 m/sec

To complete the mission design specification a detailed examination has been made of the satellite encounter viewing conditions and viewing conditions of Jupiter from these orbits. Detailed calculations of some of the satellite encounters based on the exact timing of table 2 indicated in some cases that the best encounters resulted in spacecraft impacts, and thus the timing at periapsis

had to be adjusted very slightly from that given in the table to avoid such impacts. It was found, for example, that for the 1976 opportunity the date of periapsis passage for the first encounter orbit (2.29X45.131 R_J) had to be changed by 0.016 day or 23 minutes earlier to avoid collision. The encounter characteristics with Jupiter and for each of the three satellites for the 1976 opportunity are shown in figures 11(a)-11(d). In each case, the distance from the center of each body in units of Jupiter radii is shown as a function of solar phase angle (the angle between a line from the Sun to the body and a line from the body to the spacecraft) for each of the several orbital encounters.

The orbital encounter with Jupiter is not too different for the interaction region orbits and the satellite encounter orbits. A camera system with 200-μrad resolution capability can achieve the desired 300-km resolution at a distance of about 20 R_J. At that distance the camera field of view must be about 6° to obtain full disk images. It can be seen that an average phase angle of about 60° is achieved at that distance on these orbits. Note that there is one excellent encounter with both Io and Ganymede and four encounters with Europa that more than satisfy the 10-km resolution requirement. The encounter geometries are excellent, and with about a 6° field of view nearly fully lighted disk images (phase angle 10°-20°) are obtained with 10-km resolution.

PROBLEM AREAS

The first problem area and the most critical in terms of mission accomplishment is the guidance and navigation problem associated with arriving at periapsis of the first encounter orbit on time. This must be done with very high precision. Since there are two very long period interaction region orbits prior to the first encounter orbit, a small percentage error in the interaction orbit period will result in a large error in the time of return to periapsis after the two interaction region orbits. The main sources of this error are the uncertainty in the delivered insertion impulse, the uncertainty in the periapsis radius, and the uncertainty in the planet's mass. The resulting error in the orbit period of the first interaction orbit is shown in figure 12 as a function of the error in the insertion impulse and the error in periapsis radius. It can be seen that an error of only 10 m/sec out of about 900 m/sec results in an error in the time of the next periapsis passage of about 3 days. This is, of course, completely intolerable.

Fortunately, because the orbital science requirements and the encounter timing requirements dictated the use of two high period orbits before the first satellite encounter orbits, a strategy as described below can be used to eliminate the periapsis time error. This strategy (fig. 13) involves first inserting into an orbit of 2.29 R_J periapsis by about 103 R_J apoapsis. The period of this orbit is about 5 days longer than the planned 2.29X100 R_J orbit, and thus even with the error in insertion the return to periapsis is guaranteed to be late (by as much as about 8 days). During the first orbit, however, tracking information indicates the various errors, and at the second periapsis passage apoapsis is lowered properly (to no lower than about 88 R_J) to shorten the orbit period and thus allow a return to periapsis on time after the second orbit. The error in periapsis radius is relatively unimportant relative to the satellite encounters (ref. 4), but that error can be trimmed at apoapsis of the second orbit with an impulse of less than 10 m/sec. Since the time trimming maneuver indicated above is applied at periapsis so as to always reduce apoapsis, there is no velocity penalty associated with the trimming maneuver.

Of course, the deboost maneuver after the second interaction region orbit into the first encounter orbit also has an associated error. The error in the return to periapsis of this orbit is

shown as a function of the ΔV error in figure 14. In this case, a 5 m/sec error out of about 520 m/sec results in an error in orbit period of about 0.2 day or almost 5 hr. This too is quite intolerable. However, since the best encounters do not occur until the second orbit, a subsequent trim maneuver at periapsis of the second encounter orbit can be used to eliminate this error. The penalty associated with this maneuver is no larger than the original error in the deboost maneuver.

The second problem area is associated with the high plane-change velocity requirements to achieve an equatorial orbit, particularly for the 1976 and 1977 launch opportunities. Because of this high velocity penalty, an examination was made of the possibility of eliminating the plane-change maneuver and accepting out-of-plane encounters with the satellites. Since it is possible to choose the direction of the line of nodes of the orbit plane at insertion, a study was made of the effect of various choices on the satellite encounters. It was found that the out-of-plane effects upon the encounters were nearly minimized by choosing the line of nodes to pass through the position of Io at time of encounter (fig. 15). This choice results in orbit inclination of about 7° for the middle of the launch window, and since the spacecraft is passing through the equatorial plane at the time of encounter with Io, the encounter distances with Io are the same as for the equatorial orbits.

The degradation of the encounter closest approach distances of the above nonequatorial orbits as compared with the equatorial orbits is shown in figure 16 for the 1976 launch opportunity. The open symbols indicate the closest approach distance for the equatorial orbit, while the filled symbols indicate the closest approach distance for the 7° inclined orbit. Note that the encounter distances with Io are identical for both orbits. The best approaches to Europa and Ganymede are also seen to be degraded from little over 0.1 R_J to about 1.0 R_J. This encounter distance is very marginal depending on camera optics relative to achieving a 10-km resolution on those two satellites. One advantage for the nonequatorial encounter orbit is that it is now possible to encounter Io over its north or south pole and thus to make measurements within the magnetic flux tube intersected by Io.

The final problem area to be considered here is the effect of satellite ephemeris uncertainties on the encounters. The observational limit from Earth-based telescopes is approximately 1 arc sec which translates into an uncertainty in position at a distance of 5 AU of about 3000 km. Actually, the ephemeris uncertainties are somewhat less than this amount due to long-term observation and processing. Thus, the uncertainty in position is less than 0.04 R_J, and since the closest encounter is about 0.1 R_J, the uncertainty in the satellite ephemeris does not seem too serious.

ALTERNATIVE MISSION DESIGNS

The preceding analysis has been based on the three-satellite encounter orbits defined in reference 4. It should be remembered that these encounters require not only that an orbit of a particular period (or integer multiple thereof) be established but that the periapsis radius (and hence the apoapsis radius) have certain fixed values. If the orbit is designed for repeating two-satellite encounters, rather than three-satellite encounters, the required value of the periapsis radius is relaxed and only a particular period orbit (or integer multiple thereof) is required. The analysis of the appropriate orientation and timing of periapsis passage relative to satellite alinements for such orbits is given in appendix B.

There are two primary and conflicting motives for wishing to change periapsis radius from the value associated with the three-satellite encounter orbit. The first motive is to lower periapsis to take advantage of the substantial reduction in the insertion velocity requirement for an orbit of the same period. The second, and conflicting, motive is to raise periapsis to reduce the substantial danger associated with the trapped radiation belts. Two-satellite encounter orbits possess this flexibility.

Lower Periapsis Orbits

The advantage of lowering periapsis to reduce the insertion impulse requirement is shown in figures 17(a) and 17(b), where parametrically the periapsis and apoapsis radii achievable with a given ΔV are indicated. Two values of the hyperbolic approach speed are given. Lines of constant orbit period are also shown for comparison. A mission design has been developed to typify the lowest propulsive velocity requirements consistent with the science requirements. For this mission, a periapsis radius of 1.1 R_J was chosen and no plane-change maneuver was allowed. The mission was designed for the 1976 launch opportunity, and encounters with Io and Ganymede were chosen.

Accomplishing these repeated encounters requires an orbit of the same size and with the orientation and periapsis passage relative to an Io-Ganymede alinement as calculated in appendix B and indicated in figure 18. The orbit period is 14.15 days or four times the period of Io. With a periapsis radius of 1.1 R_J the apoapsis radius must be 46.22 R_J. The alinement of periapsis must be advanced 128.16° past the direction of the alinement of Io and Ganymede. The appropriate time of periapsis passage is 0.15 day after the time of the satellite alinement. Encounters with Io and Ganymede occur just before periapsis passage.

Alinements of Io and Ganymede occur at syzygy and at two equally spaced times between syzygies. The conditions shown in figure 18 are very similar to the requirements for the mode 7 three-satellite encounters shown on the left-hand side of figure 2. The difference in the alinements shown (about 11°) and the difference in the location of periapsis relative to the approach asymptote require that the two-satellite encounter orbits be targeted for best encounters near the time of the alinement on Julian date 2443940.720. This is one syzygy period after the target alinement for the three-satellite encounter orbit. If, for convenience and to make comparisons easier, the same arrival date is chosen for this mission as in the case of the three-satellite encounter mission, approximately 7 days more must be spent on the interaction region orbits. The timing of events for this mission is given in table 4. Since the arrival date was somewhat arbitrarily set at

TABLE 4.– 1976 JUPITER ORBITER MISSION IO-GANYMEDE ENCOUNTERS, 1.1 R_J PERIAPSIS

Target alinement date	2443940.720	Mar. 8, 1979
Timing requirement for encounter	+.150	
Periapsis date for best encounter	2443940.870	Mar. 8, 1979
Two orbit periods	–28.300	
Date of entrance into 1.1X46.22 R_J orbit	2443912.570	Feb. 8, 1979
Two interaction region orbits (1.1X106.49 R_J)	–97.250	
Arrival date	2443815.320	Nov. 2, 1978

Julian date 2443815.32, the launch requirements are identical to those shown in figure 6(a) for the three-satellite encounter mission.

TABLE 5.– ΔV BUDGET – TWO-SATELLITE ENCOUNTER ORBIT, 1976 OPPORTUNITY, 1.1 R_J PERIAPSIS

Insert into 1.1X106.49 R_J orbit	610 m/sec
Periapsis rotation penalty at insertion	20
Deboost into 1.1X46.22 R_J orbit	380
Satellite perturbation trim	60
TOTAL	1070 m/sec

The total ΔV budget for the 14.15-day two-satellite encounter mission defined in table 4 is indicated in table 5. Note that no plane-change maneuver is planned and the line of nodes is positioned in the direction of Io at the time of intercept with Io, resulting in an orbit inclined 8° to the equator. The total ΔV requirement for this mission is 1070 m/sec or only about one-half the comparable requirement for the three-satellite encounter mission.

In order to complete the comparison of this alternative mission design class which attempts to minimize the orbital insertion requirements with the more ambitious three-satellite encounter mission previously defined, the encounter characteristics with Jupiter and each of the two satellites are shown in figures 19(a)-19(c). Again, distance from the center of each body is shown as a function of phase angle. The orbital encounter with Jupiter (fig. 19(a)) is very similar to that obtained on the three-satellite encounter orbit (fig. 11(a)). The average phase angle at 20 R_J (about where full disk images of the required resolution are obtained) is about 60° to 70°. Thus, about 60 percent of the disk is lighted. The resolution of IR, UV, and microwave observations is, of course, greatly enhanced near periapsis over that obtained at the higher periapsis radius.

The encounters with Io on this orbit are very good as can be seen from figure 19(b). This is to be expected since the spacecraft is passing through the equatorial plane at the time of closest approach and the orbit period was chosen as an integer multiple of the period of Io. Only planetary oblateness effects cause the encounters to degrade with time. There are at least four encounters that satisfy the resolution requirement. Up to 80 percent of the disk is seen to be lighted. Fully lighted disks are not obtained due to the inclination of the spacecraft orbit. Notice particularly the nearly symmetrical distance-phase angle relationship for the third encounter orbit. This indicates a nearly polar flyby thus satisfying the requirement to make measurements within the intersecting magnetic flux tube.

The encounters with Ganymede (fig. 19(c)) are considerably degraded as a result of out-of-plane effects. Only the second encounter orbit barely satisfies the resolution requirement and then only for a 50-percent lighted disk. The encounter is, however, nearly polar which presents the opportunity to compare the flux tube measurements for Io and Ganymede.

Higher Periapsis Orbits

The advantage of raising periapsis is to reduce to a tolerable level the damage and transient effects of the trapped radiation environment on sensitive instruments and electronic equipment. This advantage is paid for in terms of increased insertion impulse requirements as shown in figure 17. A mission design has been developed that typifies the requirements to escape at least part of the radiation hazard consistent with the science requirements and spacecraft and launch vehicle

capabilities. For this mission, a periapsis radius of 4.0 R_J was chosen and a plane-change maneuver to an equatorial orbit was required. The mission was designed for the 1976 launch opportunity and encounters with Io and Ganymede were chosen.

To accomplish the encounters, an orbit period was chosen that is a multiple of the period of Io. For orbit periods of 28.31 days the apoapsis radius must be 71.12 R_J. The appropriate location of periapsis relative to an Io-Ganymede alinement is that it must be advanced 101.52° past the direction of the alinement of the two satellites. (See appendix B.) The appropriate time of periapsis passage is 0.34 day after the time of the satellite alinement. Encounters with Io and Ganymede occur just before periapsis passage.

For the approach conditions of the 1976 launch opportunity, the high-periapsis, two-satellite encounter orbit is targeted for best encounter near the alinement of Io and Ganymede near Julian date 2443919.567. The timing of events for this mission is given in table 6. The arrival date for this mission is approximately 15 days earlier than the launch window shown in figure 6(a). This change results in a very small change in launch characteristics from those shown in that figure.

TABLE 6.– 1976 JUPITER ORBITER MISSION IO-GANYMEDE ENCOUNTERS, 4.0 R_J PERIAPSIS

Target alinement date	2443919.567	Feb. 15, 1979
Timing requirement for encounter	+.340	
Periapsis date for best encounter	2443919.907	Feb. 15, 1979
One orbit period	–28.31	
Date of entrance into 4.0X71.12 R_J orbit	2443891.597	Jan. 18, 1979
Two interaction region orbits (4.0X100 R_J)	–92.423	
Arrival date	2443799.174	Oct. 17, 1978

TABLE 7.– ΔV BUDGET – TWO-SATELLITE ENCOUNTER ORBIT, 1976 OPPORTUNITY, 4.0 R_J PERIAPSIS

Insert into 4.0X100 R_J orbit	1180 m/sec
Periapsis rotation penalty at insertion	0
Plane-change to equatorial orbit	570
Deboost into 4.0X71.12 R_J orbit	220
Satellite perturbation trim	60
TOTAL	2030 m/sec

The total ΔV budget for the high periapsis two-satellite encounter mission defined in table 6 is indicated in table 7. Notice that the plane-change maneuver requirement for the 4.0X100 R_J orbit is similar to the requirement for the 2.29 R_J orbit. The total ΔV requirement for this mission is 2030 m/sec or about the same as the requirement for the three-satellite encounter mission.

The viewing conditions for Jupiter and the two satellites are shown in figures 20(a)-20(c). It can be seen that the average phase angle at 20 R_J (full disk images) is about 45°. Thus about 75 percent of the disk is lighted. The resolution at periapsis is, of course, considerably degraded over what is obtained from the lower periapsis orbits.

The encounters with Io on this orbit are very good as can be seen from figure 20(b). The spacecraft approaches Io from out of the Sun and does a terminator flyby. Full disk images are

obtained with average phase angles from about 10° to about 45° on the first four passes. The single encounter with Ganymede on the first orbit (fig. 20(c)) is very good, coming within about 0.1 R_J. Full disk images with an average phase angle of 30° are obtained.

MISSION DESIGN SUMMARY

The relatively specific mission designs outlined in the preceding sections vary over a fairly wide range of required capability and have varying degrees of scientific accomplishment. The actual mission designs chosen are not too important. It has been demonstrated that considerable flexibility exists in the mission design, particularly in the satellite encounters. Thus the actual missions described herein can be modified considerably as the science objectives and the instrument capabilities are examined in greater detail. The purpose of this paper is to lay some of the groundwork for the final mission design effort by indicating some of the many options available.

Table 8 indicates the span of the mission requirements for the three classes of missions discussed in previous sections. These three classes are: (1) the three-satellite repeated encounter orbits; (2) the low-periapsis, two-satellite repeated encounter orbits; and (3) the high-periapsis, two-satellite repeated encounter orbits. The second class are nonequatorial low-inclination orbits, while the first and third class are equatorial orbits. The total insertion, plane change (if any), deboost and orbital trim requirements are given in the table for missions during the 1976, 1977, and 1981 opportunities. Each mission has launch window characteristics identical or very nearly so to those shown in figures 6(a)-6(c). Broken plane heliocentric transfers have not been considered in this table but, as indicated before, this technique could save about 200 m/sec for the class (1) and (3) orbits for the 1976 opportunity. The total impulsive velocity required in each case for the 1977 opportunity is higher than that required for the 1976 opportunity for orbit classes (1) and (3) because of a higher plane-change requirement. It is also higher for orbit class (2) because of a greater requirement for periapsis rotation. The requirements for orbit classes (1) and (3) for the 1981 opportunity are lower than the comparable requirements in the other years in spite of a higher

TABLE 8.– TOTAL ΔV REQUIREMENTS

Orbit period, day	Orbit size, R_J	ΔV Requirement, m/sec		
		1976	1977	1981
Orbit class (1) three-satellite encounter orbit, equatorial				
7.111	2.391X27.483	2750	3010	2590
14.222	2.290X45.131	2020	2280	1920
21.333	2.255X59.884	1780	1980	1670
Orbit class (2) low-periapsis two-satellite encounter orbit, nonequatorial				
7.08	1.1X28.72	1470	1600	1850
14.15	1.1X46.22	1070	1140	1360
21.23	1.1X60.90	890	950	1150
49.54	1.1X107.98	660	700	880
Orbit class (3) high-periapsis two-satellite encounter orbit, equatorial				
28.31	4.0X71.12	2030	2160	1920
49.54	4.0X105.08	1780	1910	1730

approach velocity and hence higher insertion requirements, because of the much lower plane-change requirement for that opportunity. Orbit class (2), however, has the highest requirement during the 1981 opportunity due to the higher approach speed and higher periapsis rotation requirements.

The values given in table 8 can be changed slightly by altering the mission design, but for the most part they indicate to within less than 100 m/sec the on-board propulsion requirement for various Jupiter orbiter missions. It can be seen that the values range from about 700 m/sec to almost 3000 m/sec, but the majority of mission designs have a requirement for 2000 m/sec or slightly less. It therefore appears that if a spacecraft were designed with a propulsive capacity of about 2000 m/sec, a great deal of mission design flexibility would be available.

MISSION STRATEGIES FOR SEVERE RADIATION HAZARD

In this section several mission design strategies are presented and discussed in an attempt to explore more completely the degree of mission selection flexibility available with an on-board propulsive capability of 2000 m/sec. In these strategies it is assumed that the radiation belt uncertainty is so severe as to force the initial periapsis radius to 6 R_J, which is higher than previously discussed. Initial on-board measurement and exploration of the belts from the outermost fringes downward is assumed to be required to achieve a reasonable probability of mission success.

Parametric data showing what can be accomplished by propulsive maneuvers at periapsis are given in figures 17(a) and 17(b), which indicate the apoapsis radius for establishing an orbit from hyperbolic approach as a function of periapsis radius and the value of the propulsive retromaneuver. For example, slightly over 1400 m/sec is required to established a 6X100 R_J orbit. In addition, the effect of subsequent periapsis maneuvers can be determined by the addition of that increment to the propulsive requirement to establish the initial orbit. For example, if 200 m/sec is used for retro at periapsis of a 6X100 R_J orbit, a 6X75 R_J orbit results. For reference, lines of constant orbit period are indicated by the dash-dot lines.

Figure 21 indicates what can be accomplished by propulsive maneuver at apoapsis The final periapsis resulting from an apoapsis retromaneuver of a given amount applied at an apoapsis radius of 100 R_J is shown as a function of the initial periapsis radius. For example, 400 m/sec applied at apoapsis is required to change a 6X100 R_J orbit to a 3X100 R_J orbit.

With the use of figures 17 and 21 several strategies utilizing a total propulsive capability of 2000 m/sec have been developed. Each strategy is discussed in detail. These particular strategies have been developed for the 1976 launch opportunity, but with slight modifications they could be used for any opportunity. In all cases, only nonequatorial orbits were considered to increase the degree of flexibility.

The first strategy is similar to mission profiles previously discussed in that an initial interaction region orbit is established of about 6X100 R_J followed by a subsequent reduction in apoapsis and orbit period to about 21 days for satellite encounters. This, of course, assumes that the perceived radiation hazard at 6 R_J is deemed too high to proceed lower. With a periapsis of 6 R_J it is still possible to encounter the satellite Io but such an encounter occurs at periapsis which is undesirable due to interference with observations of Jupiter at that time. For this reason, it is suggested that encounters with Europa and Ganymede be considered. The timing and velocity requirements for

this mission are shown in tables 9 and 10, respectively. It can be seen that this mission design requires just about 2000 m/sec of on-board propulsion. The viewing conditions of Jupiter and the satellites are shown in figures 22(a)-22(c).

TABLE 9.– 1976 JUPITER ORBITER MISSION EUROPA-GANYMEDE ENCOUNTERS, 6.0 R_J PERIAPSIS

Target alinement date	2443908.994	Feb. 4, 1979
Timing requirement for encounter	–.570	
Periapsis date for best encounter	2443908.424	Feb. 3, 1979
One orbit period	–21.310	
Date of entrance into 6.0X56.10 R_J orbit	2443887.114	Jan. 13, 1979
Two interaction region orbits (6.0X100 R_J)	–95.102	
Arrival date	2443792.012	Oct. 10, 1978

TABLE 10.– ΔV BUDGET – TWO-SATELLITE ENCOUNTER ORBIT, 1976 OPPORTUNITY, 6.0 R_J PERIAPSIS

Insert into 6.0X100 R_J orbit	1430 m/sec
Periapsis rotation at insertion	50
Deboost into 6.0X56.10 R_J orbit	510
Satellite perturbation trim	60
TOTAL	2050 m/sec

The second strategy assumes that after establishing an orbit of 6X100 R_J (requiring 1430 m/sec) the perceived radiation hazard is low enough to reduce periapsis to, say, 4 R_J. This requires a total apoapsis retromaneuver of 250 m/sec. The orbit apoapsis is subsequently reduced and the period lowered to about 28 days for satellite encounters. The timing and velocity requirements for this mission design are shown in tables 11 and 12. The total velocity requirement for this mission is also just about 2000 m/sec. In this case, encounters with Io and Ganymede were

TABLE 11.– 1976 JUPITER ORBITER MISSION IO-GANYMEDE ENCOUNTERS, 6.0 R_J PERIAPSIS REDUCED TO 4.0 R_J

Target alinement	2443919.567	Feb. 15, 1979
Timing requirement for encounters	+.34	
Periapsis date for best encounters	2443919.907	Feb. 15, 1979
One orbit period (4.0X71.12 R_J)	–28.310	
Date of entrance into 4.0X71.12 R_J orbit	2443891.597	Jan. 18, 1979
One-half orbit period (4.0X100 R_J)	–23.106	
Apoapsis date	2443868.491	Dec. 25, 1978
One orbit period (5.0X100 R_J)	–46.880	
Apoapsis date	2443821.611	Nov. 9, 1978
One-half orbit period (6.0X100 R_J)	–23.776	
Arrival date	2443797.835	Oct. 16, 1978

TABLE 12.– ΔV BUDGET – TWO-SATELLITE ENCOUNTER ORBIT, 1976 OPPORTUNITY, 6.0 R_J PERIAPSIS REDUCED TO 4.0 R_J

Insert into 6.0X100 R_J orbit	1430 m/sec
Periapsis rotation at insertion	0
Apoapsis retro into 5.0X100 R_J orbit	120
Apoapsis retro into 4.0X100 R_J orbit	130
Deboost into 4.0X71.10 R_J orbit	220
TOTAL	1900 m/sec

planned. The viewing conditions of Jupiter are almost the same as shown in figure 20(a). The viewing conditions of the satellites are shown in figures 23(a) and 23(b).

The third strategy assumes that the exploration of the belts is of great scientific and programmatic importance. In this case an initial orbit of 6X103.26 R_J is established and so timed to have satellite encounters with Europa and Ganymede. In addition, the interaction region experiments are carried out. The spacecraft is then committed to exploration of the radiation belts in any number of steps desired until a periapsis radius of 2.0 R_J is reached. The timing and velocity requirements for this mission design are shown in tables 13 and 14. The total velocity requirement is, of course, 2000 m/sec. The viewing conditions of Jupiter are similar to those shown in figure 22(a). The viewing conditions of the satellites are shown in figures 24(a) and 24(b).

TABLE 13.– 1976 JUPITER ORBITER MISSION EUROPA-GANYMEDE ENCOUNTERS, 6.0X103.26 R_J ORBIT

Target alinement	2443908.994	Feb. 4, 1979
Timing requirement for encounters	-.550	
Periapsis date for best encounters	2443908.444	Feb. 3, 1979
Two orbit period	-99.420	
Arrival date	2443809.024	Oct. 27, 1978

TABLE 14.– ΔV BUDGET – TWO-SATELLITE ENCOUNTER ORBIT, 1976 OPPORTUNITY, 6.0X103.26 R_J ORBIT

Insert into 6.0X103.26 R_J orbit	1400 m/sec
Periapsis rotation at insertion	20
Total apoapsis retromaneuvers down to 2.0 R_J	580
TOTAL	2000 m/sec

These are not all the strategies that are possible with a spacecraft with about 2000 m/sec of on-board propulsive capability, but they do demonstrate the range of flexibility. It also appears that most experimental objectives can be satisfied to some degree.

CONCLUSIONS

It has been shown that a single orbiting spacecraft with an appropriately designed mission profile can provide varying observational requirements necessary for some understanding of: (1) the interaction of Jupiter with the solar media; (2) the character of the major regular satellites of Jupiter; (3) the character of the near Jovian environment; and (4) the planetological and meteorological phenomena of Jupiter.

The on-board propulsive requirement for such missions range from about 700 to 3000 m/sec, but most of the possible mission design options have a requirement very near 2000 m/sec. Each mission design considered has a varying degree of accomplishment associated with the science

objectives. In general, however, each mission discussed above provides at least five passes through the interaction region; provides encounters of at least two satellites with multiple encounters of at least one satellite at distances which allow photography with resolutions of at least 10 km; provides at least 10 orbital maps of the field and particle environment surrounding Jupiter; and provides synoptic observations of Jupiter over a range of wavelengths and various degrees of photographic coverage with resolutions of 300 to 30 km.

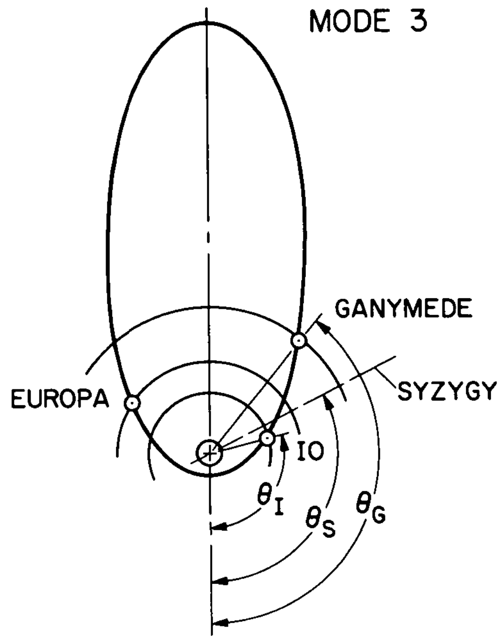
The greatest uncertainty associated with feasibility of a Jupiter orbiter mission is the trapped radiation environment, which can have damaging effects upon the sensors and the electronics of the spacecraft. Several mission strategies have been developed that meet the science objectives stated above; these missions require about 2000 m/sec of propulsive capability and have a high probability of escaping the effects of the radiation environment.

National Aeronautics and Space Administration
Moffett Field, Calif., 94035, Feb. 14, 1972

APPENDIX A

ORIENTATION AND TIMING FOR THREE-SATELLITE ENCOUNTERS

In addition to having the appropriate size orbit for satellite encounters, it is necessary to have the location of periapsis oriented properly and to be at periapsis at the appropriate time all with respect to the position and time of syzygy. This location and time of periapsis passage can be determined from the true anomalies of intercepts given in reference 4.



Sketch (a).– Mode 3 encounter geometry.

Sketch (a) shows the positions at which mode 3 encounters occur. Table 15 gives the locations of intercept from reference 4 for the three different size mode 3 orbits of interest. With the positions and hence the times of satellite intercepts relative to periapsis given, the required position and time of syzygy relative to the periapsis passage conditions can be calculated as follows. The time spent by the spacecraft between intercepts with Io and Ganymede can be equated to the time spent by Io moving from intercept to syzygy plus the time spent by Ganymede moving from syzygy to intercept:

$$t_G - t_I = \frac{\theta_S - \theta_I}{W_I} + \frac{\theta_G - \theta_S}{W_G}$$

where W_I and W_G are the mean motions of the satellites Io and Ganymede, respectively. Therefore, the true anomaly of syzygy is given by:

$$\theta_S = \left(t_G - t_I + \frac{\theta_I}{W_I} - \frac{\theta_G}{W_G}\right) \frac{W_I W_G}{W_G - W_I}$$

The time of periapsis passage relative to the time of syzygy is then given by

$$t_s = -\left(t_I + \frac{\theta_S - \theta_I}{W_I}\right)$$

These two relationships have been used to calculate the appropriate position and time of periapsis relative to the syzygy alinement for the orbits of interest. See table 16.

TABLE 15.– MODE 3 INTERCEPT CONDITIONS

Orbit period, days	7.111	14.222	21.333
Intercept true anomaly, deg			
Io	107.88	107.04	106.72
Europa	–129.34	–126.45	–125.47
Ganymede	147.13	141.62	139.86

TABLE 16.– ORIENTATION AND TIMING FOR MODE 3 ENCOUNTERS

Orbit period, days	7.111	14.222	21.333
True anomaly of syzygy, deg	118.74	117.13	116.53
Time of periapsis passage relative to syzygy, day	–.263	–.249	–.243

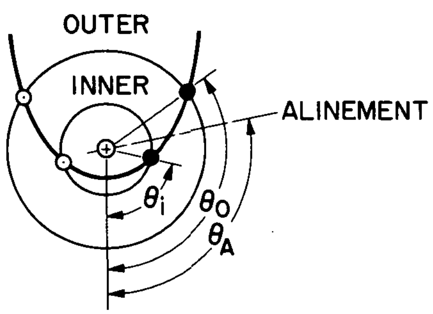
For the mode 7 encounter situation, which is the mirror image of mode 3 (Europa is encountered after periapsis and Ganymede and Io before periapsis), the true anomaly of syzygy and the time of periapsis passage relative to the time of syzygy are equal in value but opposite in sign to the values for mode 3.

APPENDIX B

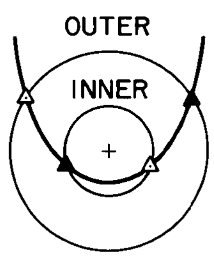
ORIENTATION AND TIMING FOR TWO-SATELLITE ENCOUNTERS

In the establishment of an orbit about a planet, the orbit is determined by the time of periapsis insertion, periapsis radius, period, orientation of the line of apsides, inclination, and the line of nodes. Where three satellite encounters are desired (appendix A), all six of these parameters are constrained. The satellites must have nearly the same inclination if multiple close encounters at all three are desired. In the case of Io, Europa, and Ganymede the inclination is zero degrees. The period is allowed to vary in integer multiples of approximately Ganymede's period. The time of arrival, periapsis radius, and line of apsides are then determined from the encounter geometry. In the case of two-satellite encounters, the elimination of the third satellite frees one parameter, the radius of periapsis. Therefore, the periapsis radius can be selected on the basis of propulsion, hazard avoidance, or some other criteria, being limited only in that the maximum value must be inside the inner of the two satellites to be encountered.

MODES 1 & 2



MODES 3 & 4



MODE	SYMBOL	SIGN OF θ_i	SIGN OF θ_o	FIRST ENCOUNTER
1	●	+	+	INNER
2	○	-	-	OUTER
3	▲	-	+	OUTER
4	△	+	-	INNER

Sketch (b).– Two-satellite encounter geometry.

The possible encounter modes are shown in sketch (b). There are three, two-satellite combinations of interest: Io-Europa, Io-Ganymede, and Europa-Ganymede. As with the three-satellite encounters, it is convenient to use the alinement of the satellites (in this case when the two satellites of interest are at the same planetocentric longitude) as a reference point from which to determine the periapsis passage time t_A relative to the time of the satellite alinement and the required orientation of the alinement relative to periapsis θ_A.

Following the notation shown in sketch (b), the necessary conditions for repeated encounters are:

$$R_J < r_p < r_i \qquad \tau = n\tau_i$$

where τ is the orbital period and the subscripted quantities r_i and τ_i are the radius and period, respectively, of the innermost satellite of the pair to be encountered. The variable n is a positive integer. From the above equation, the semimajor axis of the spacecraft orbit is given by:

$$a = r_i n^{2/3}$$

and hence the eccentricity by

$$e = (a - r_p)/a$$

The true anomaly of the satellite intercept can be calculated from the equations

$$\cos\theta_i = \left\{[a(1 - e^2)/r_i] - 1\right\}/e$$

$$\cos\theta_o = \left\{[a(1 - e^2)/r_o] - 1\right\}/e$$

where the subscripts i and o stand for the inner and outer satellites, respectively.

With the positions and hence times of intercept relative to periapsis on the spacecraft orbit determined, the analysis proceeds as in the case of the three-satellite encounter orbit presented in appendix A. As shown in sketch (b), the time spent by the spacecraft between intercepts of the inner and outer satellites is equated to the time spent by the inner satellite in moving from intercept to the alinement at position θ_A plus the time spent by the outer satellite moving from the alinement to intercept

$$t_o - t_i = (\theta_A - \theta_i)/W_i + (\theta_o - \theta_A)/W_o$$

where W_i and W_o are the respective mean motions of the satellites. Solving for the position of the alinement relative to periapsis yields

$$\theta_A = [t_o - t_i + (\theta_i/W_i) - (\theta_o/W_o)]\,[W_iW_o/(W_o - W_i)]$$

The time of periapsis passage relative to the time of the satellite alinement is then given by

$$t_A = -[t_i + (\theta_A - \theta_i)/W_i]$$

By the appropriate choice of sign for θ_i and θ_o, the orientation and timing of the four encounter sequences as shown in sketch (b) can be calculated. For example, table 17 indicates the orientation and timing requirements for Io-Ganymede encounters for modes 1 and 3, and for two values of periapsis radius and orbit period. The requirements for modes 2 and 4 are mirror images of 1 and 3, respectively, and are obtained by simply changing sign on the angle θ_A and the time t_A.

TABLE 17.– ENCOUNTERS OF IO AND GANYMEDE

Mode	Period τ, days	Radius of periapsis r_p, R_J	True anomaly of Io θ_i, deg	True anomaly of Ganymede θ_o, deg	True anomaly of alinement θ_A, deg	Periapsis passage time relative to alinement time t_A, day
1	14.15	1.1	131.9	154.0	128.2	-0.15
		4.0	73.3	128.0	103.6	-.35
3	14.15	1.1	-131.9	154.0	191.7	-1.42
		4.0	-73.3	128.0	125.5	-.78
1	42.46	1.1	130.3	151.0	127.3	-.15
		4.0	71.1	122.1	100.8	-.33
3	42.46	1.1	-130.3	151.0	190.4	-1.41
		4.0	-71.1	122.1	122.6	-.77

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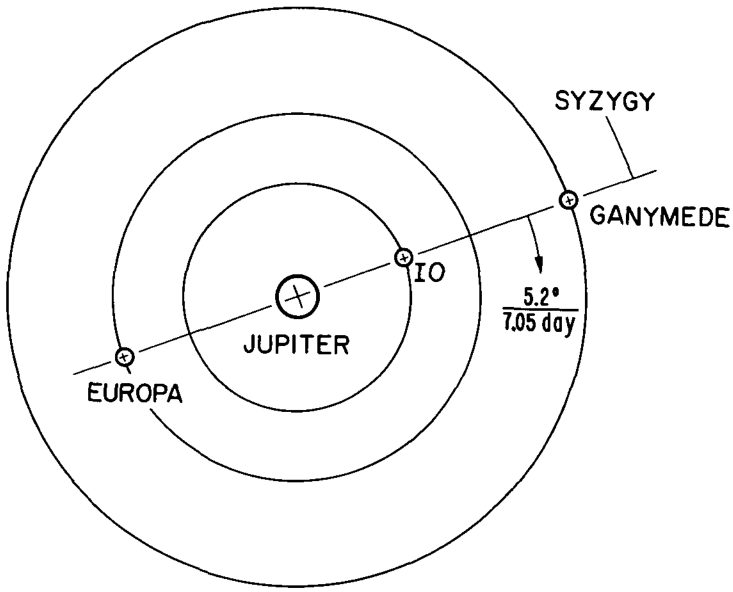


Figure 1.– Satellite alinement.

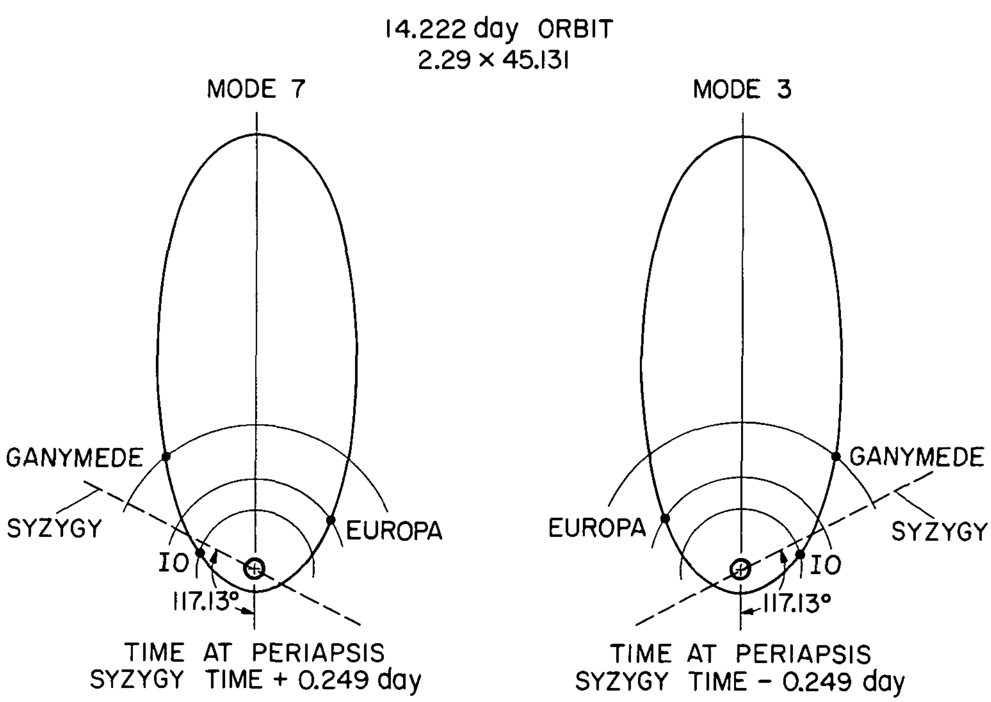
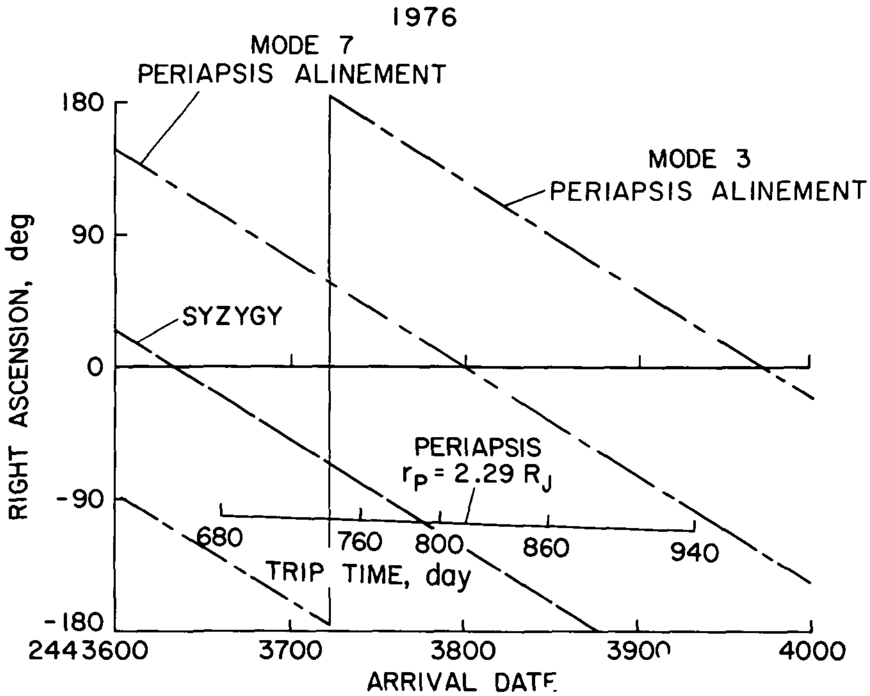
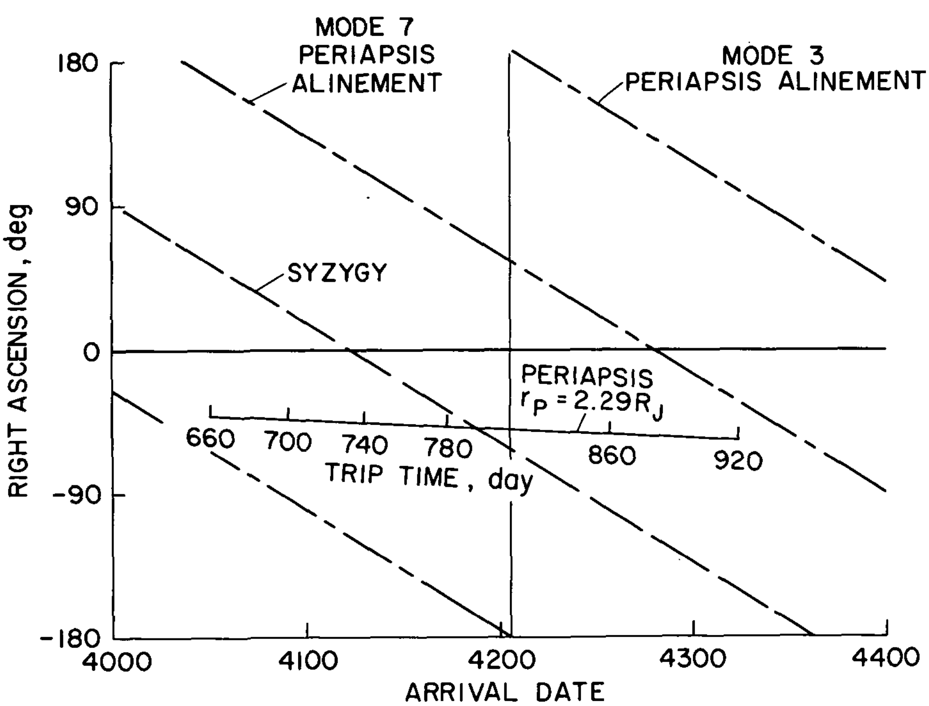


Figure 2.– Three-satellite encounter requirements.

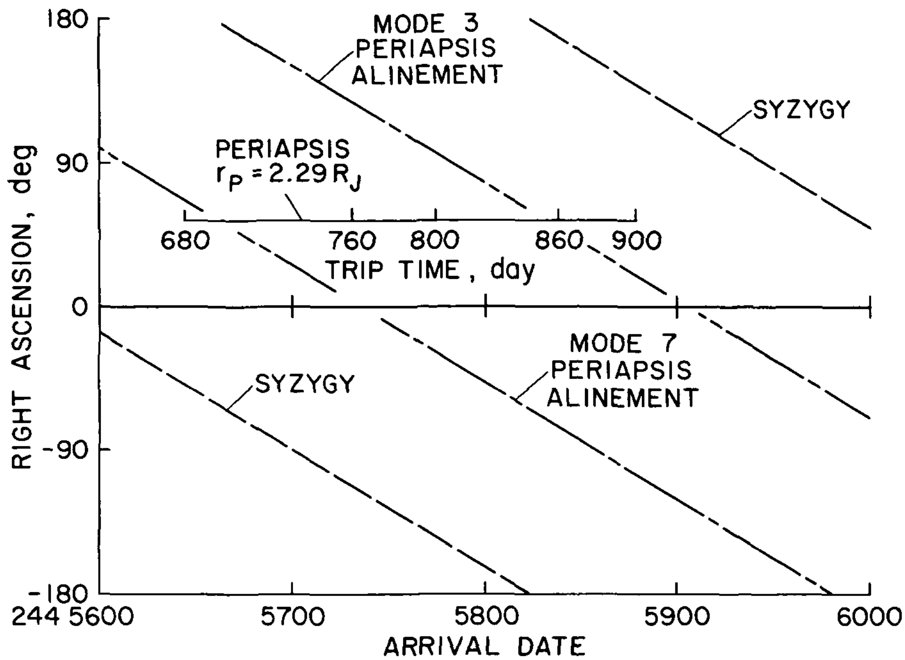


(a) 1976



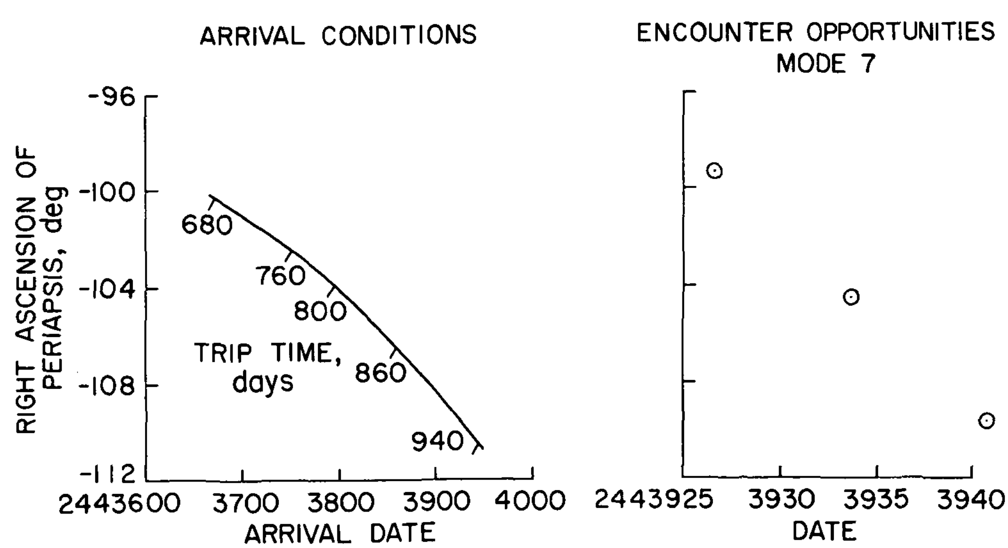
(b) 1977

Figure 3.– Encounter alinement.



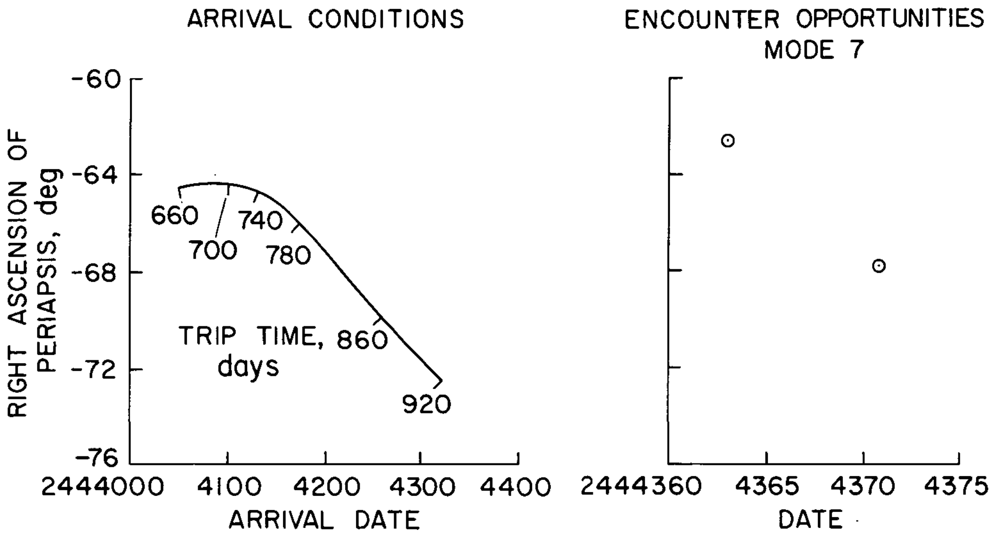
(c) 1981

Figure 3.– Concluded.

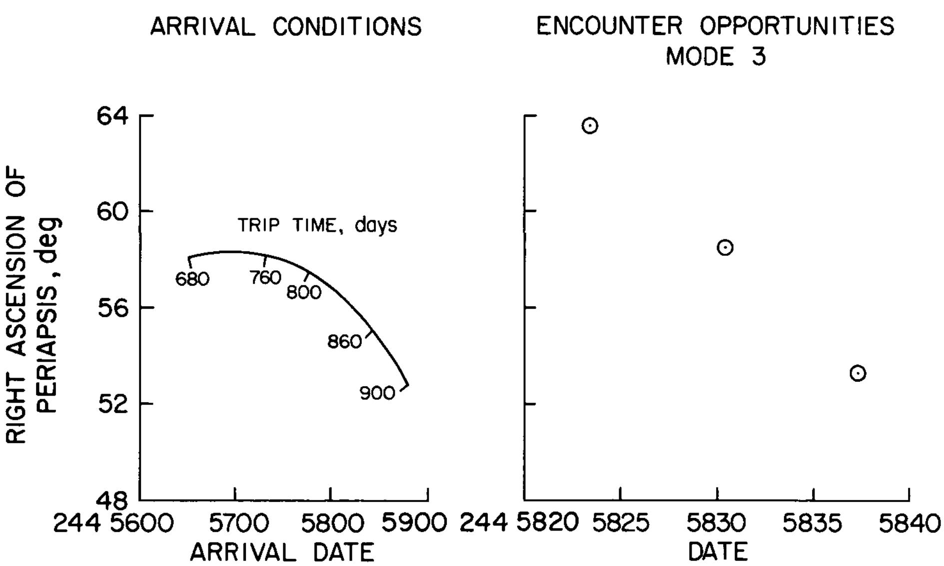


(a) 1976

Figure 4.– Encounter opportunities.



(b) 1977



(c) 1981

Figure 4.— Concluded.

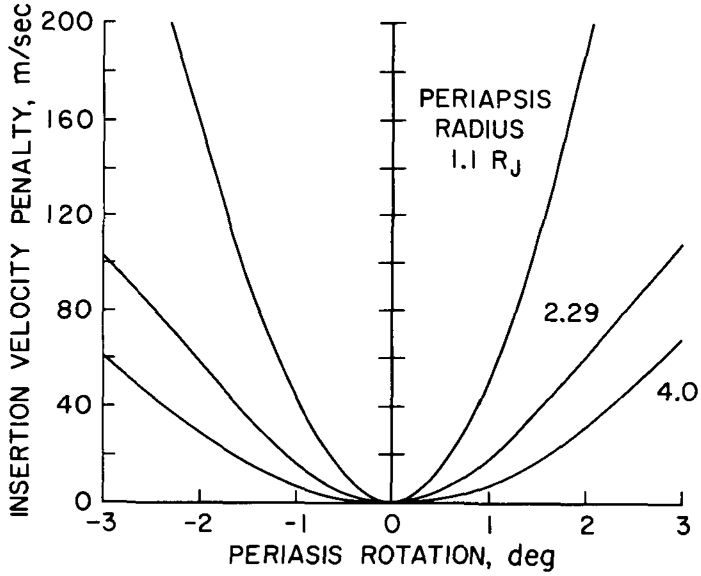
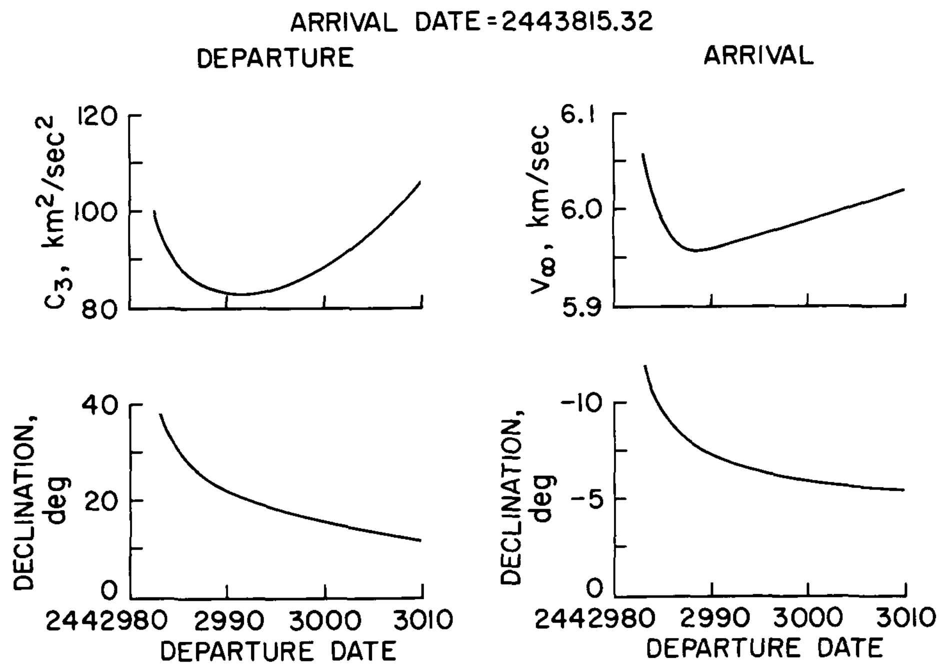


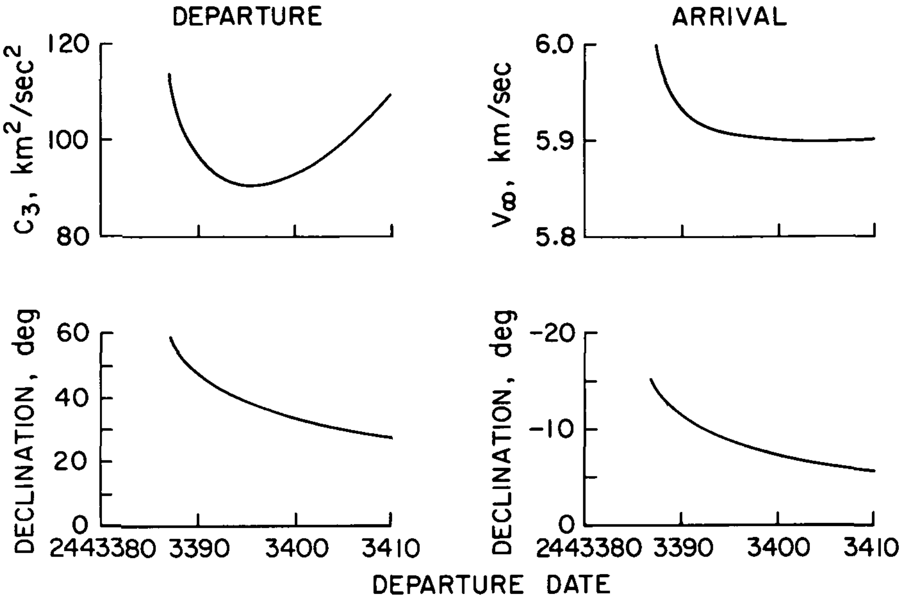
Figure 5.– Off-periapsis insertion penalty.



(a) 1976

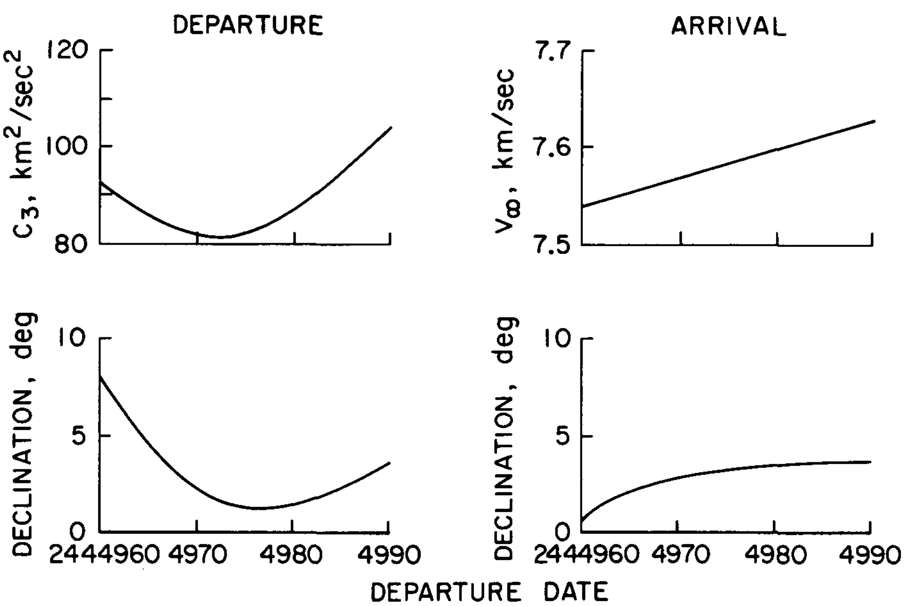
Figure 6.– Launch window characteristics.

ARRIVAL DATE = 2444252.478



(b) 1977

ARRIVAL DATE = 2445711.521



(c) 1981

Figure 6.– Concluded.

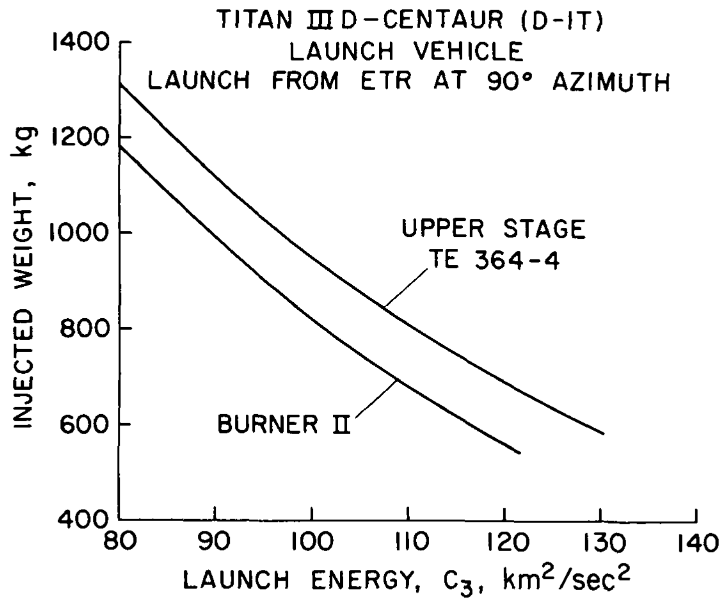


Figure 7.— Launch vehicle capability.

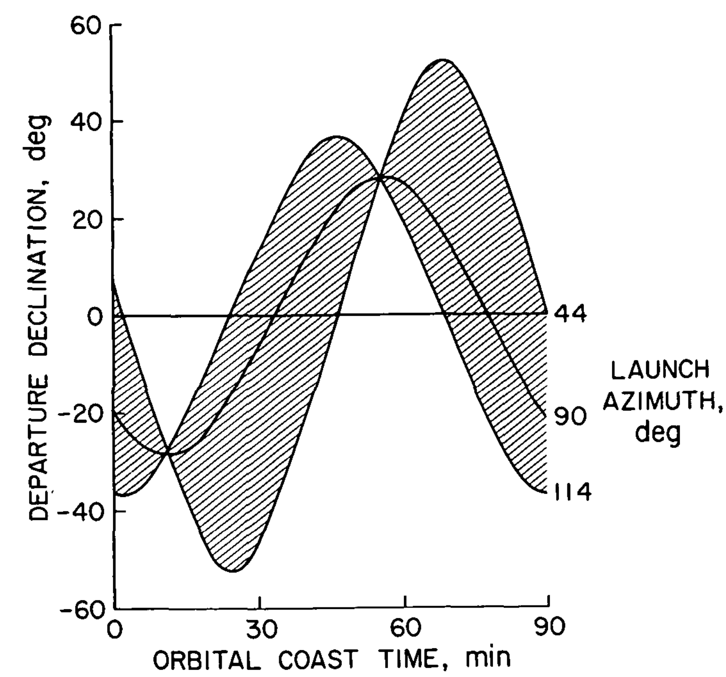


Figure 8.— Orbital coast time requirements for departure.

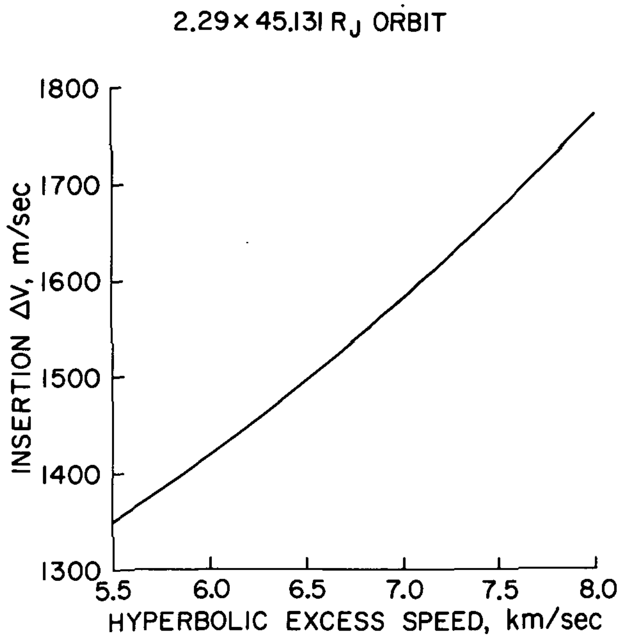


Figure 9.– Jupiter orbit insertion velocity requirements.

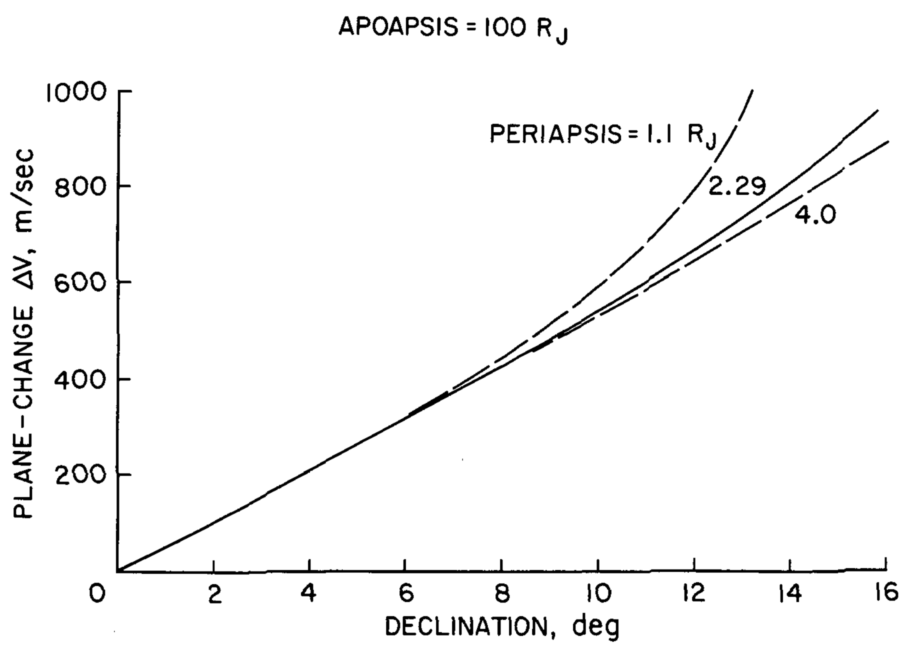
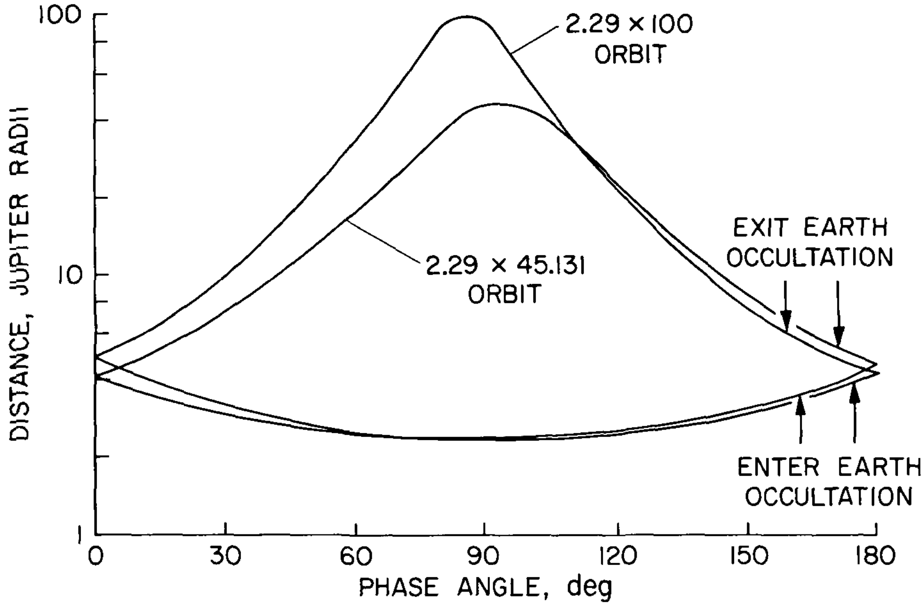
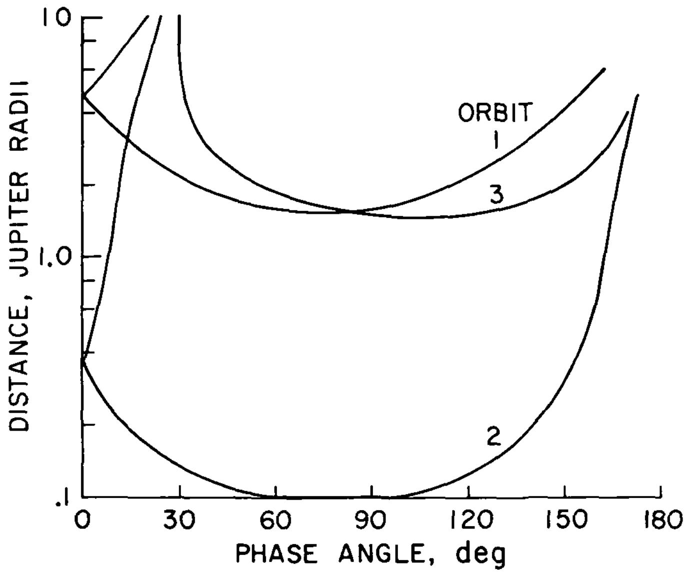


Figure 10.– Plane-change velocity requirements to achieve equatorial orbits.

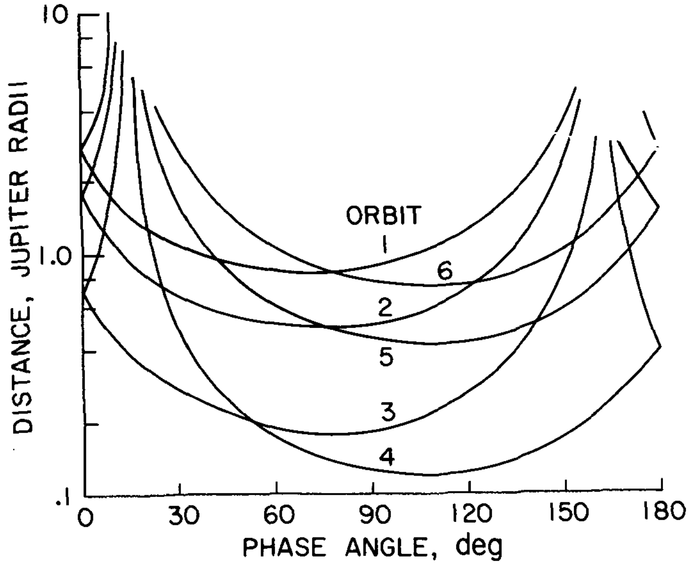


(a) Jupiter.

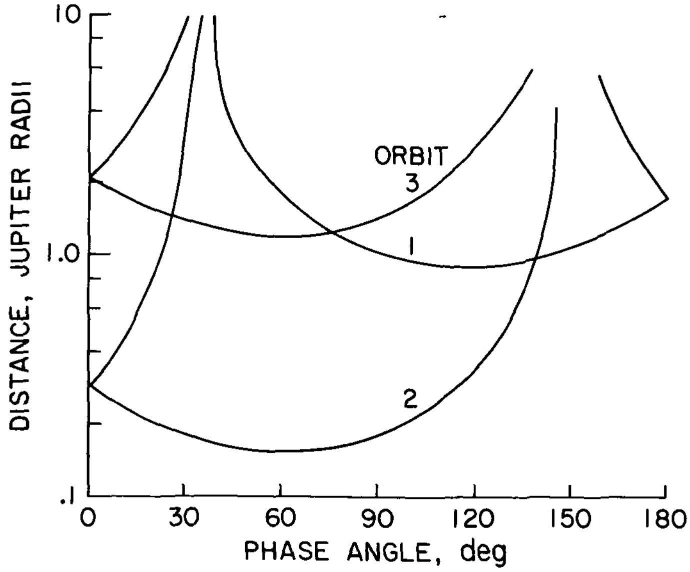


(b) Io.

Figure 11.– 1976 encounter characteristics.



(c) Europa.



(d) Ganymede.

Figure 11.– Concluded.

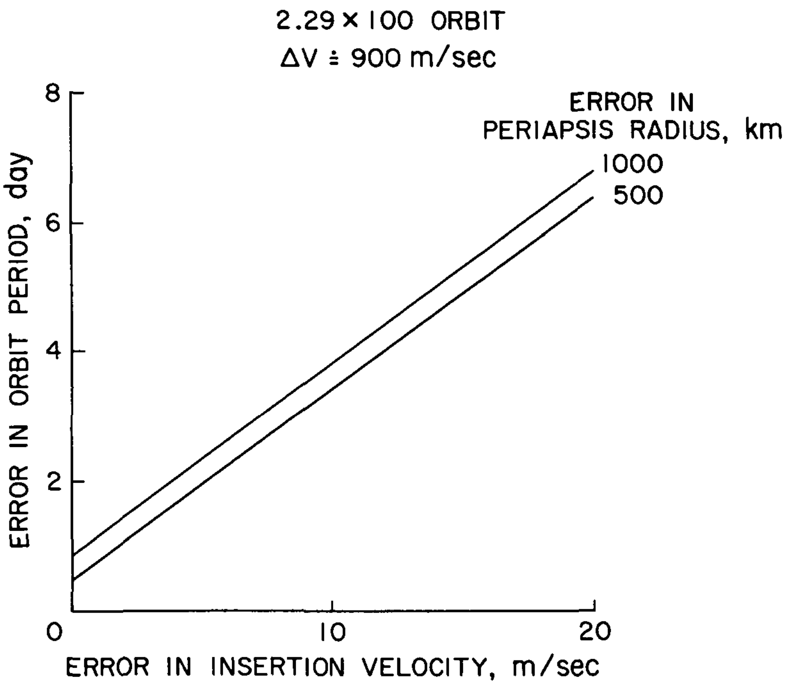


Figure 12.— Periapsis-time error after first interaction orbit.

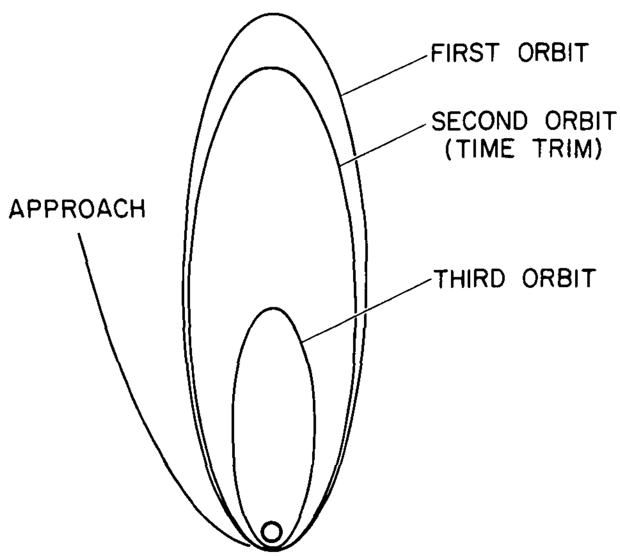


Figure 13.— Periapsis-time trimming schematic.

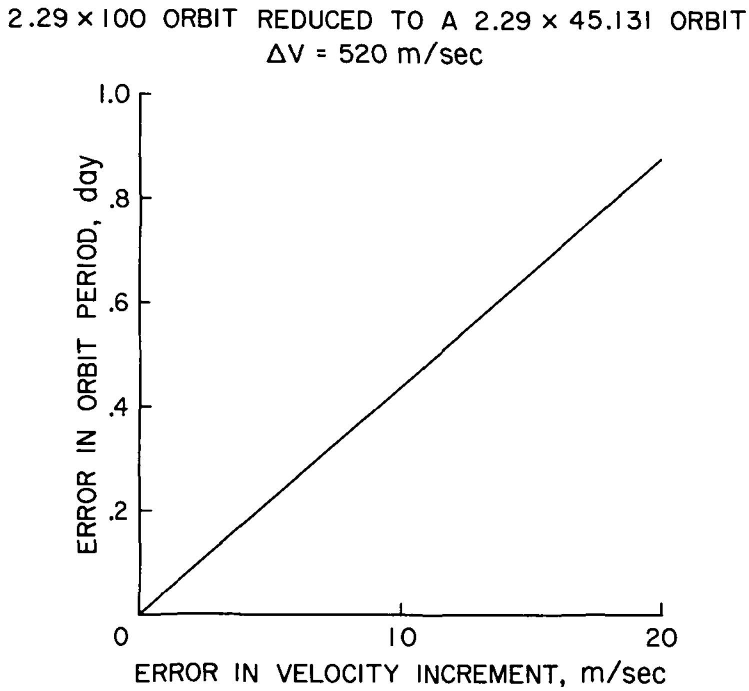


Figure 14.– Periapsis-time error after first encounter orbit.

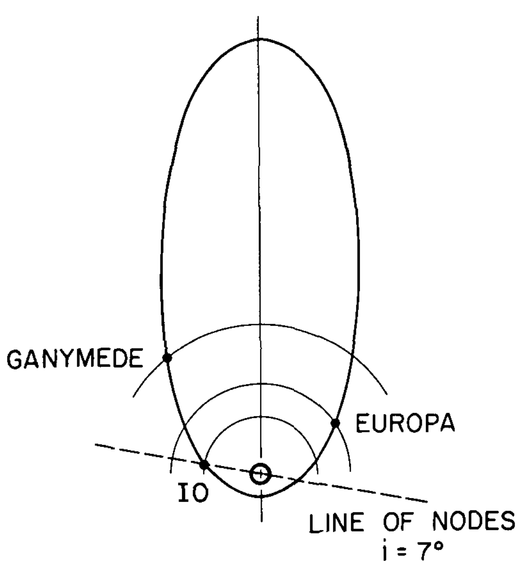


Figure 15.– Minimizing out-of-plane effects – 1976.

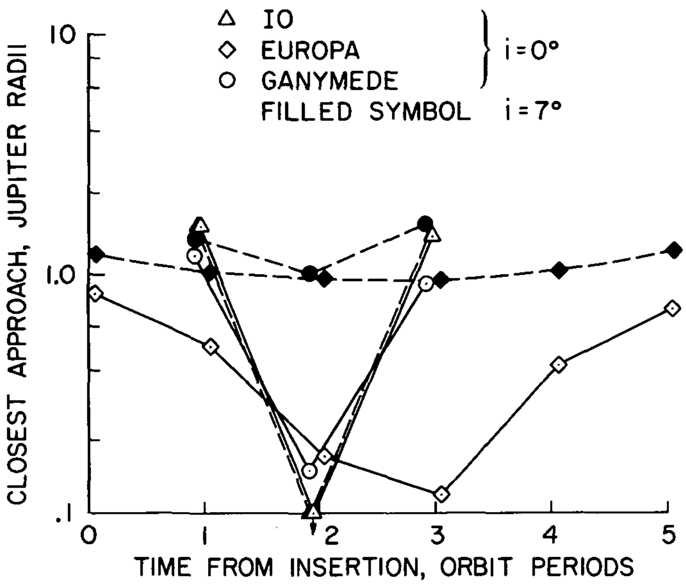
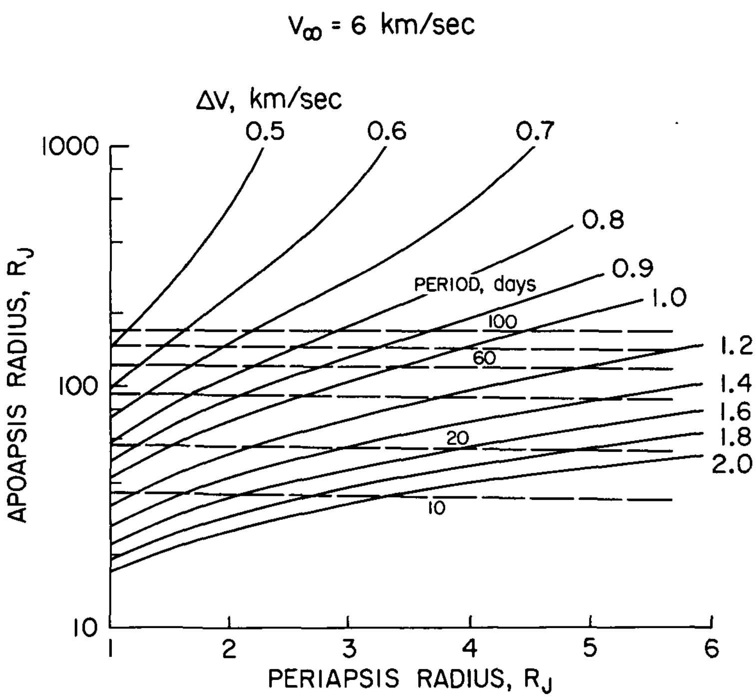
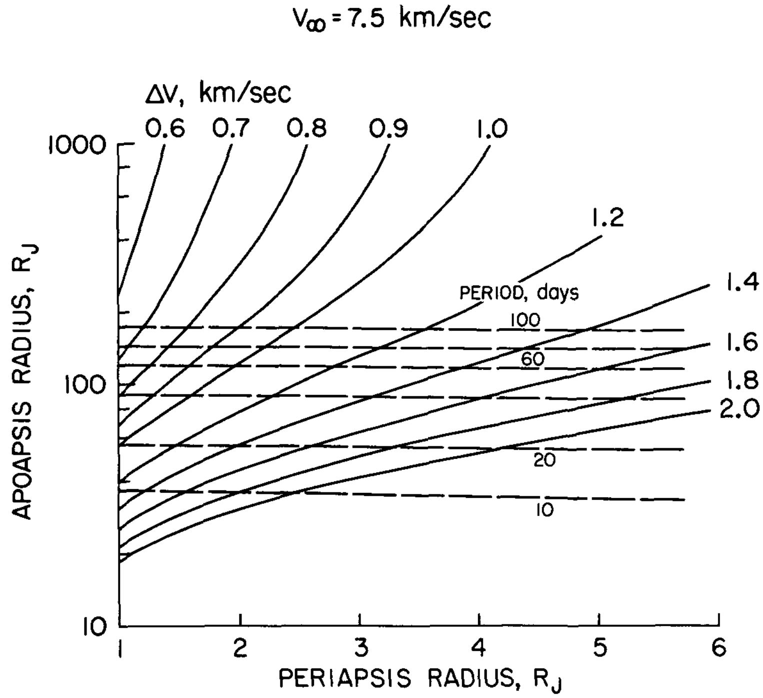


Figure 16.– Encounter characteristics for nonequatorial orbits – 1976.



(a) V_∞ = 6 km/sec

Figure 17.– Orbit characteristics and insertion requirements for orbits about Jupiter.



(b) V_∞ = 7.5 km/sec

Figure 17.– Concluded.

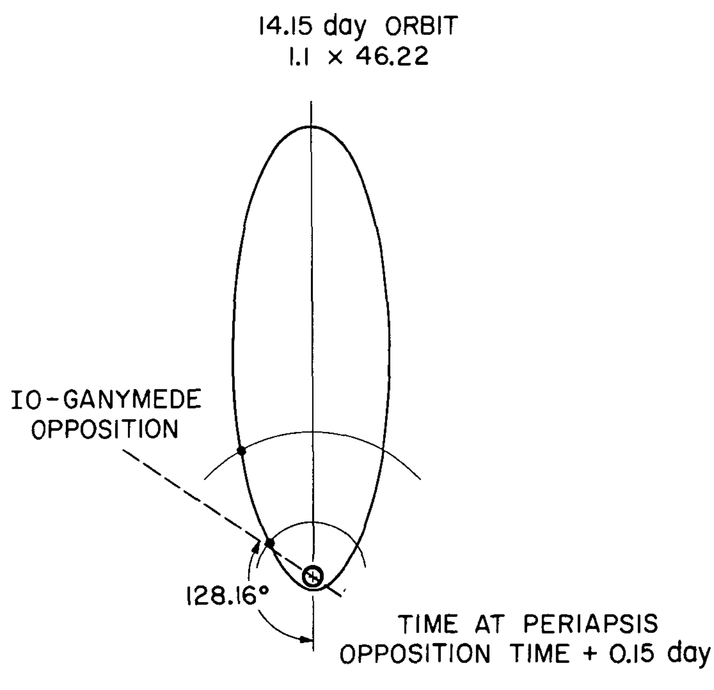
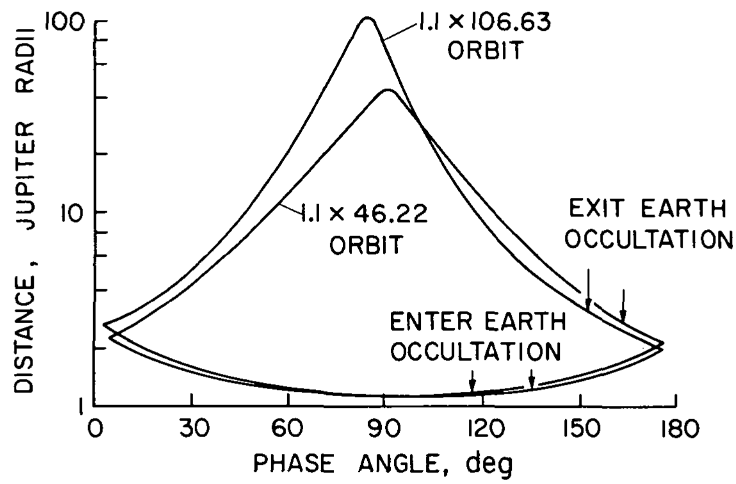
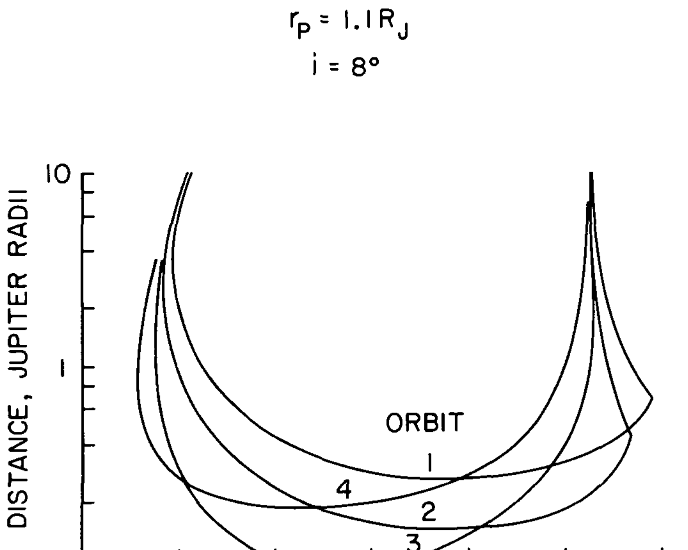


Figure 18.– Two-satellite encounter requirements.

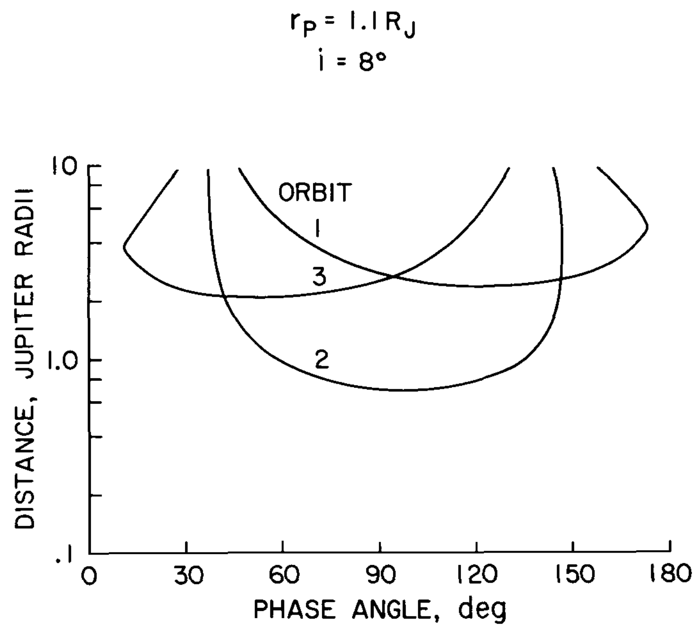


(a) Jupiter.



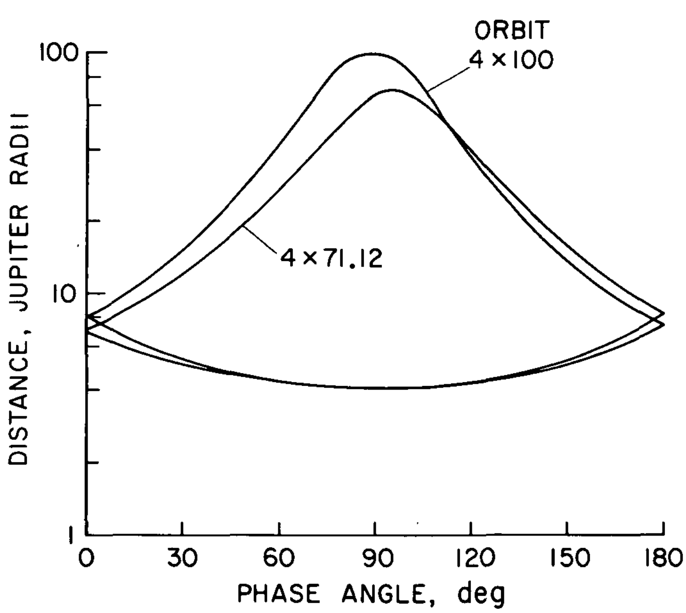
(b) Io.

Figure 19.– 1976 encounter characteristics (r_p = 1.1 R_J).



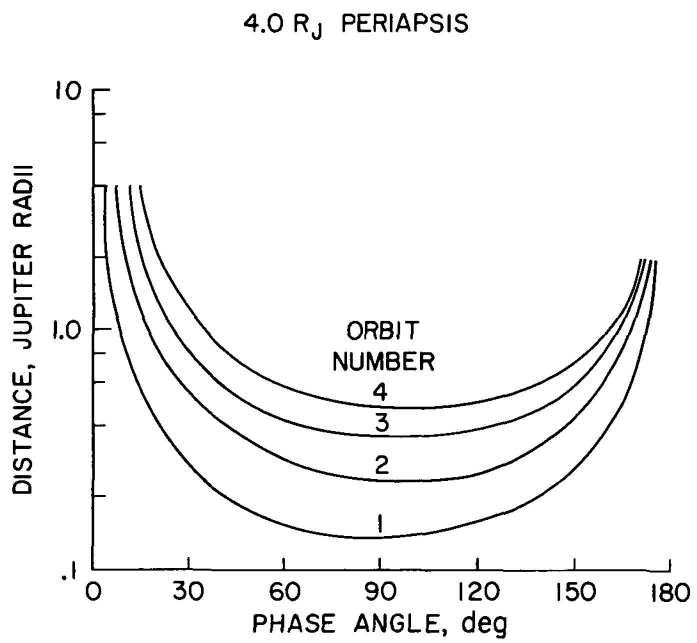
(c) Ganymede.

Figure 19.– Concluded.

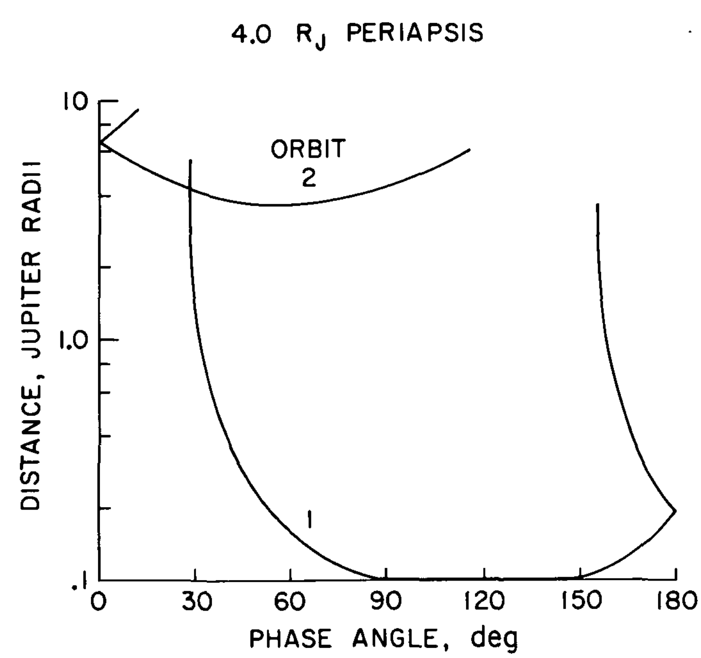


(a) Jupiter.

Figure 20.– 1976 encounter characteristics (r_p = 4.0 R_J).



(b) Io.



(c) Ganymede.

Figure 20.– Concluded.

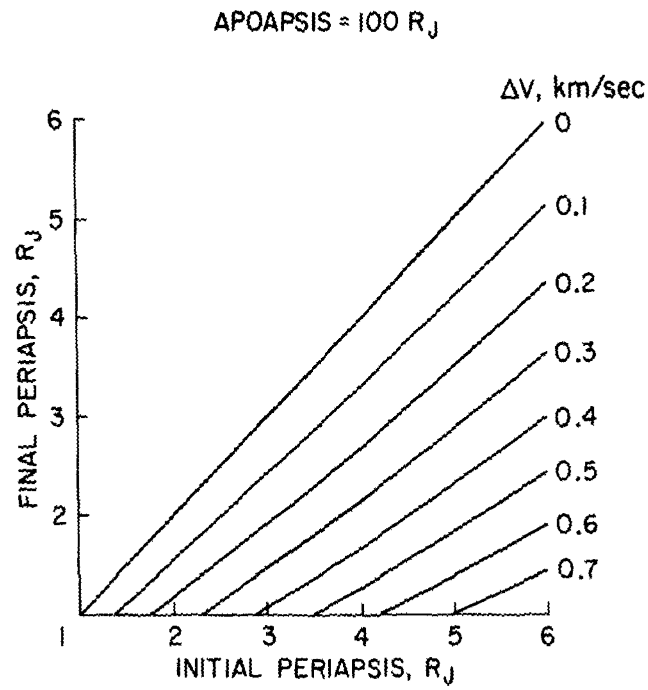
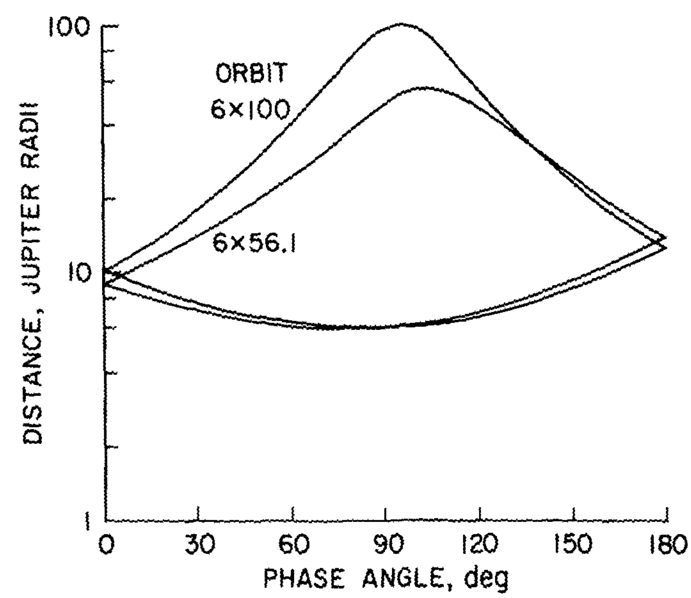


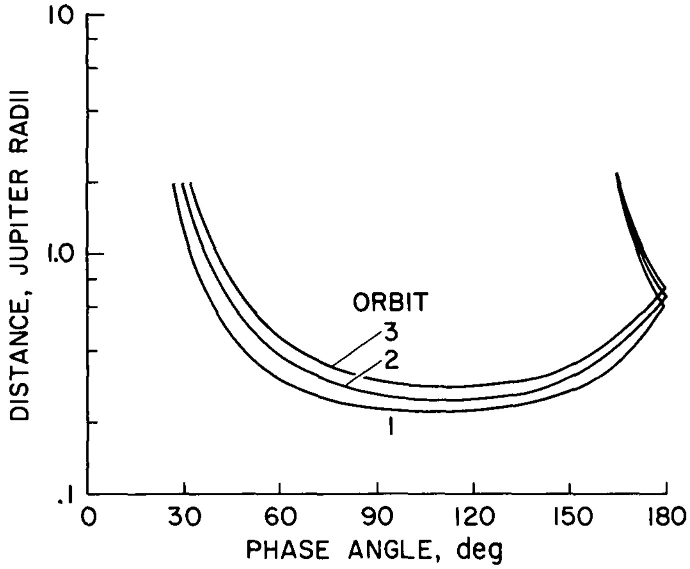
Figure 21.– Periapsis modifications via apoapsis retromaneuvers.



(a) Jupiter.

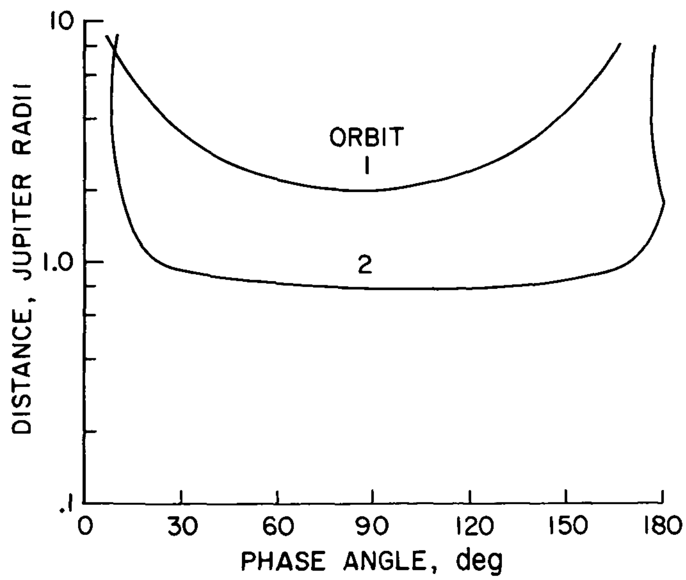
Figure 22.– 1976 encounter characteristics ($r_p \approx 6.0\ R_J$).

6.0×56.1 R_J ORBIT



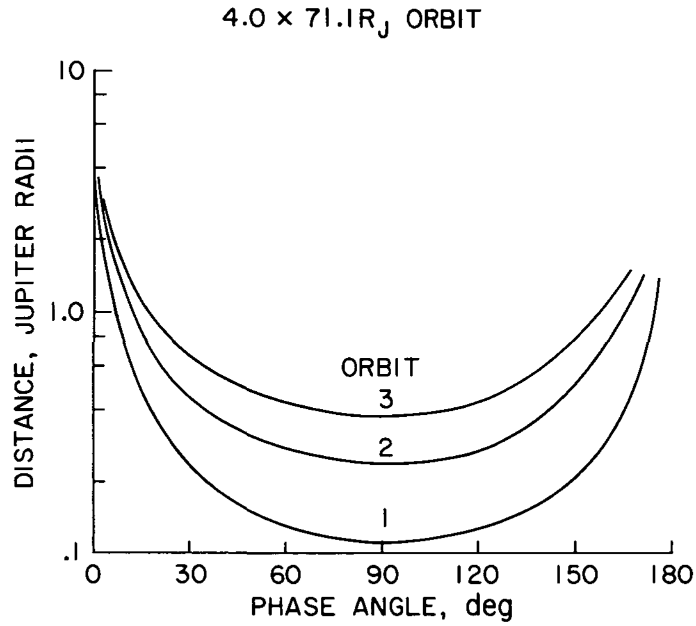
(b) Europa.

6.0×56.1 R_J ORBIT

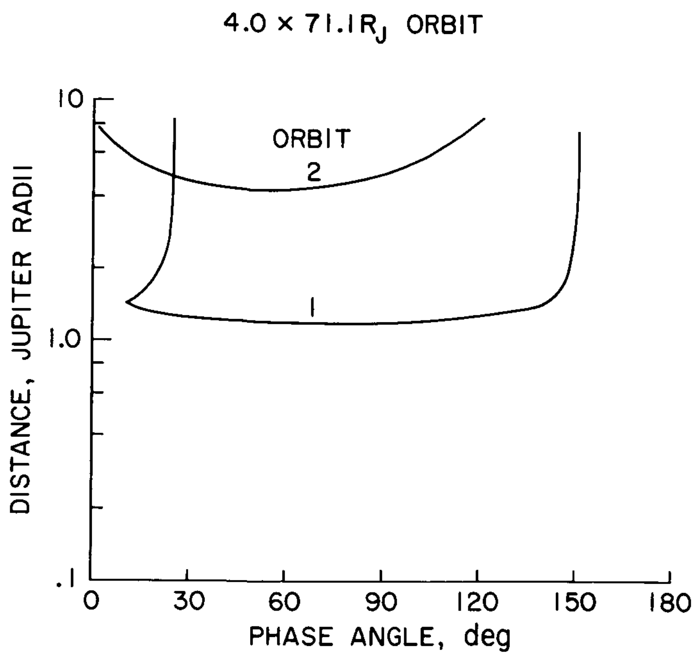


(c) Ganymede.

Figure 22.– Concluded.

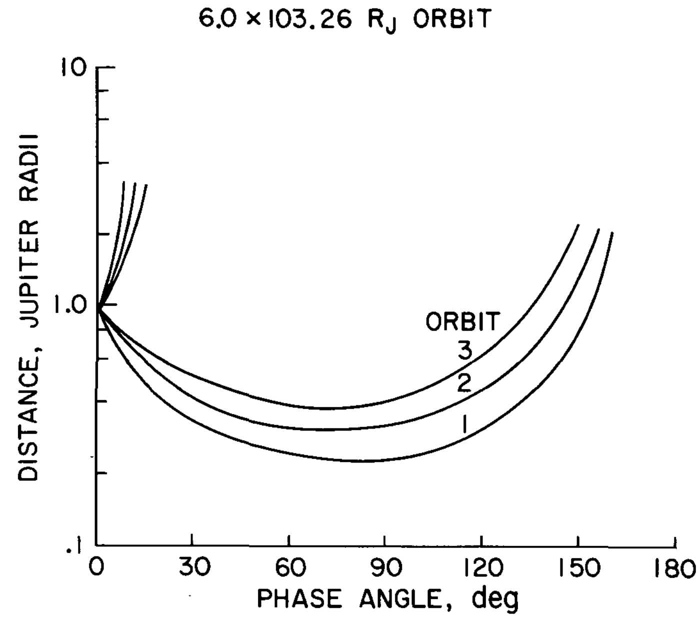


(a) Io.

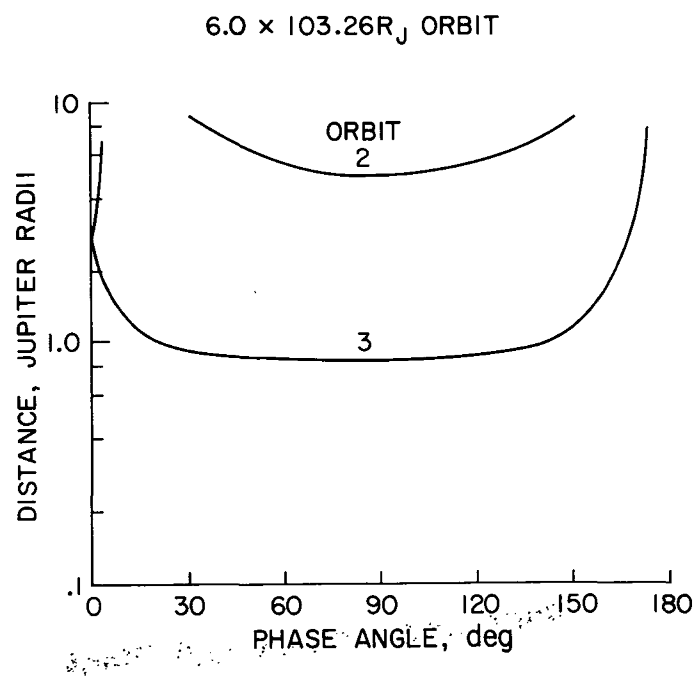


(b) Ganymede.

Figure 23.– 1976 encounter characteristics (r_p = 6.0 R_J reduced to 4.0 R_J).



(a) Europa.



(b) Ganymede.

Figure 24.– 1976 encounter characteristics (6.0 X 103.26 R_J orbit).

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